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NASA Contract NAS 5-302

PROJECT APOLLO

A Feasibility Study of an Advanced Manned Spacecraft and System

FINAL REPORT

VOLUME IV. ON-BOARD PROPULSION

Book 1 — Text and Appendix P-C

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Contract NAS 5-302

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GENERAL ELECTRIC

MISSILE AND SPACE VEHICLE DEPARTMENT

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CHAPTER I

ON-BOARD PROPULSION

I. ON-BOARD PROPULSION

1.0 Summary

Propulsion for the APOLLO spacecraft has been studied in breadth and depth. The breadth of study encompasses detailed parametric analyses by the General Electric Missile and Space Vehicle Department and a number of leading rocket propulsion companies in the United States. This study, then, analyzes proposed propulsion systems in depth and evaluates specific details of engine design, construction, operation, and qualification requirements.

From this basic study, the proposed APOLLO propulsion system has been synthesized. It is potentially capable of providing safe and reliable propulsion for the lunar orbiting mission, the most demanding of the APOLLO missions. This powerplant will, then, be capable of providing propulsion for many less demanding, earlier missions and is adequate to meet super-orbital abort requirements.

In addition, the presence of a crew demands a design that is highly reliable and as nearly fail-proof as possible. From detailed design studies conducted by the Aerojet-General Corporation, Bell Aerosystems, and the Reaction Motors Division of the Thiokol Chemical Corporation, it appears that the mission requirements are best served by a pressurized hydrogen-oxygen powerplant as designed by Aerojet-General.

Their design, AJ-10-133, incorporates a high degree of redundancy on such key elements as the thrust chambers and tankage. Thus, it provides a high level of reliability. Further, their design approach incorporates proven state-of-the art technology and components accrued from the Hydra/Hylas Program. This will enable the system to be fully developed by 1963 for earth orbital and rendezvous missions with a booster smaller than the Saturn C-2. Design flexibility and growth assure success in developing a 1966 system capable of performing circumlunar and lunar orbital flights.

For booster abort, it is recommended that a multiple unit solid propellant system be incorporated. This type of system will result in a high level of reliability and will minimize the weight penalty associated with the abort system. The solid abort and separation motors of Thiokol's Elkton Division are suggested.

The attitude control requirements are best met by contemporary earth-storable propellant reaction control system as designed by the Marquardt Corporation.

2.0 Introduction

The APOLLO propulsion system study has been conducted by the General Electric Company's Missile and Space Vehicle Department in association with the leading rocket propulsion companies. Solid propellant designs were prepared by the Elkton Division of Thiokol Chemical Corporation and Aerojet-General Corporation's Solid Rocket Plant. Attitude control studies were made by Bell Aerosystems, Aerojet-General, the Reaction Motors Division of Thiokol, and the Marquardt Corporation. The main on-board propulsion system designs were prepared by Bell Aerosystems, Reaction Motors Division of Thiokol, and the Liquid Rocket Plant of Aerojet-General. Finally, an analysis of fuel energy management was made by the Astronautics Corporation of America.

The Missile and Space Vehicle Department has a unique capacity to evaluate propulsion systems. Having no vested interest in propulsion systems as a product, this department has weighed the various propulsion schemes strictly on a merit basis for application to the APOLLO vehicle.

Specifications were prepared for the various subcontractors to ensure system integration of the propulsion system with the APOLLO vehicle. A two-phase development was envisioned. It was considered that an early system, available in the 1963 time period, would be apropos for earth orbital missions and rendezvous. These early missions will serve for vehicle development, crew training, and will supply further information relative to prolonged periods of weightlessness, etc. Such data will be useful in the progression to the eventual circumlunar and lunar orbital flights. Thus, propulsion design was directed toward a 1963 vehicle with growth capability to the eventual 1966 vehicle. The premise is that power plant modifications and improvements for the 1966 vehicle will be achieved without major revisions of the 1963 system. The complete initial specification given to the propulsion subcontractors is presented in Table I-2-I. The specifications have been prepared such that the resulting designs are applicable to either the D-2, the semi-ballistic re-entry vehicle finally selected, or to the R-3 modified lenticular re-entry vehicle.

Throughout this study, emphasis has been placed upon safety and reliability as essential features of the propulsion system. Utilization of state-of-the-art technology has been emphasized. It is felt that the APOLLO vehicle warrants a powerplant specifically designed for the mission. While the General Electric Company has examined units under development, such as Nomad and Centaur, the compromises associated with using these systems probably outweigh any potential savings in cost.

A pictorial representation of the breadth of this study is presented in Figure I-2-1.

Presented in this volume are the details of the Missile and Space Vehicle Department's recommended approach to the propulsion for APOLLO. The final-study results of the six subcontractors are presented as separate appendices to this volume.

TABLE I-2-I. INITIAL APOLLO PROPULSION DESIGN SPECIFICATIONS

Design Parameters		D-2	R-3
A. Abort			
1.	Initial Abort g's (in the direction of thrust)	20	15
2.	Burning time, seconds, for 6 units	1.0	1.9
	for 2 units	2.0	-
3.	Aborted weight (exclusive of Abort Propulsion) lb, 1963	7,000	6,000
	1966	6,500	5,500
4.	Number of Abort units	8	6
5.	Units dropped at end of first stage	4	4
6.	Units dropped at end of second stage	2	0
7.	Abort Rocket Angle (mounting), degrees	25	20 (Avg.)
8.	Net thrust vector through abort C.G., degrees off vertical	15	20
B. Vehicle Weight			
9.	Total vehicle weight, lb 1963	15,715	-
	1966	14,715	-

TABLE I-2-I. INITIAL APOLLO PROPULSION DESIGN SPECIFICATIONS (Continued)

Design Parameters		
C. Attitude Control (Over-all Vehicle)	Total Vehicle*	Re-entry Vehicle*
10. Total Impulse, pound-seconds	60,000	7,000
11. Maximum number of starts	3,000	500
12. Maximum single impulse, pound-seconds	200	100
13. Unit thrust, pounds	3	18
14. Number of units	12	4
15. Location of units	above numbers calculated on the basis of 9-foot lever arms.	
D. Mid-Course Correction (Outbound)		
16. Outbound ΔV, feet/second	250	-
17. Minimum g's	.25	-
18. Maximum g's	1.5	-
19. Number of starts (maximum)	5	-
E. Entering Lunar Orbit		
20. Required ΔV, feet/second	3,500	-
21. Minimum g's	.25	-
22. Maximum g's	1.5	-
23. Number of starts	2-4	-
F. Leaving Lunar Orbit		
24. Required ΔV, feet/second	3,500	-
25. Minimum g's	.33	-
26. Maximum g's (approximate)	2	-
27. Number of starts (maximum)	2	-

*Both D-2 and R-3

TABLE I-2-I. INITIAL APOLLO PROPULSION DESIGN SPECIFICATIONS (Continued)

Design Parameters	Total Vehicle*	Re-entry Vehicle*
G. Mid-Course Correction (Inbound)		
28. Inbound ΔV, feet/second	250	-
29. Minimum g's (approximate)	.5	-
30. Maximum g's (approximate)	3	-
31. Number of starts (maximum)	5	-
H. General		
32. Mission Duration, days	14	-
33. Type of Design	1963 System 1966 System	- -
I. Propulsion Design Parameter		
34. Number of thrust chambers	4	-
35. Gimbal Angle, any direction, degrees	±5	-
36. Fuel Tank Compartments, minimum	2	-
37. Oxidizer Tank Compartments (may be separate spheres), minimum	2	-
38. Propellant Reserves, percent	10	-
39. Residual Propellant (Outage) percent	3	-
40. Boil-Off Allowance		
a. Assuming non-vented pressurized tanks, percent	0	-
b. Assuming a pumped system	As required	-
J. Solid Separation Rockets (Semi-Ballistic Vehicle Only)		
41. Number of Units	4	-
42. Burning time (approximate) seconds	0.75	-
43. Unit Thrust (each motor) Pounds (approximate)	11,000	-

*Both D-2 and R-3

TABLE I-2-I. INITIAL APOLLO PROPULSION DESIGN SPECIFICATIONS (Continued)

Desired Output Information

A. Drawings

1. Powerplant schematics
2. Powerplant installed in vehicle
3. Details of important or unique components
4. Method of Tankage Compartmenting
5. Pertinent dimensions

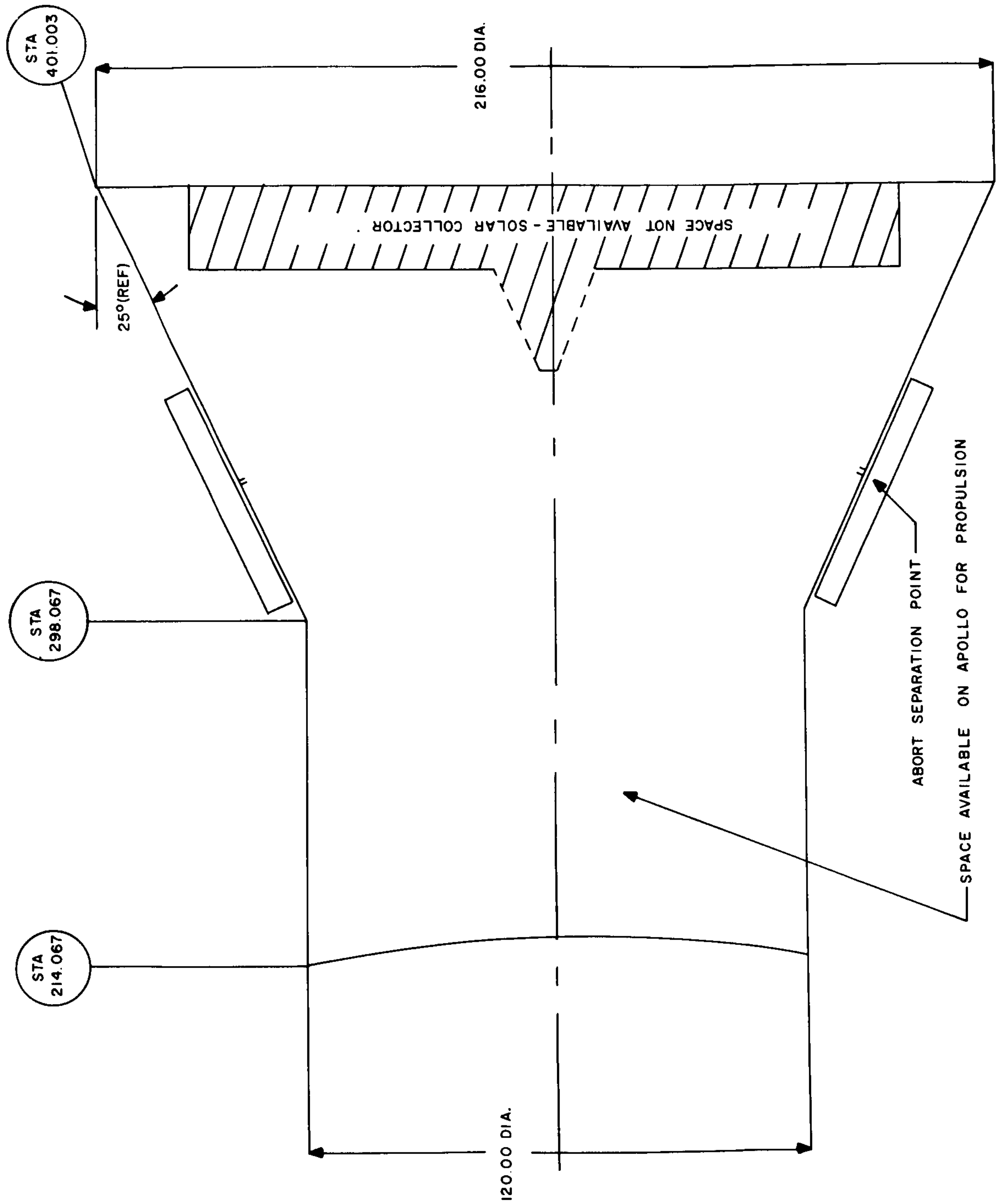
B. Technical Data

1. Weights, breakdown by major components
2. Operating parameters such as chamber pressure, thrust, expansion ratio, specific impulse, etc.
3. Reliability level expected
4. Trade-off studies made in selecting chamber pressure, expansion ratio, etc.
5. Data on alternate systems studied, such as different propellants, etc.
6. Discussion of critical items such as pressurization techniques, pumps if used, flow control, two-phase flow operation on starting and running
7. Applicable experience with propellants, components, etc.
8. Heating analysis (heat input to propellant tanks)
9. Provision for system redundancy
10. For liquid system: Predictability of start and shutdown transients, i.e., deviations from normal

    For solid system: Curves of thrust vs time at Sea Level
11. Cockpit display parameters

TABLE I-2-I. INITIAL APOLLO PROPULSION DESIGN SPECIFICATIONS (Continued)

C. Maximum Envelope for Power Plant



	Abort	Midcourse, Lunar Orbit	Attitude Control
GE - MSVD	Systems Management		
Bell Aerosystems	———	Pumped/Pressurized F_2H_2	Study
Aerojet-General	Study (Solid Rocket Plant)	**Pressurized H_2/O_2 "Hylas"**	Study (Liquid Rocket Plant)
Thiokol Chemical Corporation	**Study (Elkton)**	Pressurized MMH/N_2O_4 (1963) Pressurized MMH/OF_2 (1966) (Reaction Motors Division)	Study (RMD)
Marquardt Corp.	———	———	**Study**

Shaded Areas Are General Electric's Recommendations

Figure I-2-1. APOLLO propulsion study

3.0 Safety and Reliability

3.1 GENERAL PHILOSOPHY

The key to successful accomplishment of the APOLLO mission lies in a realistic approach to providing both safe and reliable propulsion. The solution to these problems lies not in any one single area such as engineering, design, test, or quality control. Rather, it is necessary that both safety and reliability be emphasized as an intimate ingredient at each step of design, development, cost, and operation.

Safety and reliability of propulsion are required for a number of distinguishable phases (defined in Section 4.2) of the APOLLO mission, including:

Boost (from pad to orbital velocity):	phases 1, 2, and 3
Escape:	phase 4
Space Maneuvers:	phases 5 and 6
Return to earth:	phases 7, 8, 9, and 10

During boost phases, safety alone is the dominant factor. It is obviously impractical to qualify the Saturn C-1 and C-2 vehicles for manned flight. So, a rationale has emerged that manned vehicles be capable of rapid removal from danger areas of the booster by the prompt application of abort thrust.

This in turn dictates that the abort system, of necessity, must be both reliable and qualified for manned use. Safety is achieved through use of multiple, highly reliable abort motors. Development of this boost abort reliability is discussed in Section 5.0.

During super-orbital escape, safety of the crew is again paramount, and reliability of the main propulsion system is essential for prompt return of the APOLLO capsule. In this configuration, the complete engine must operate successfully, providing over 1 g of acceleration for maneuvering capability to ensure safe return of the crew in minimum time. Safe operation of the powerplant is of primary importance, since safe shutdown could be made of part or all of the main propulsion with safe, but slower, return of the capsule.

In space maneuvers, phases 5 and 6 of propulsion, safety is essential. Damage of the space capsule for any reason is unreconcilable, and every precaution will be employed to ensure minimum possibility of damage from propulsion malfunction. Proper programming of the mission can assure that the APOLLO can always return safely to earth and propulsion can, therefore, be discontinued at any time.

It is in that portion of the APOLLO mission related to lunar orbit and return-to-earth, phases 7 through 10, that the dual importance of both safety and reliability of propulsion becomes evident. A rocket engine which shuts down safely following malfunction indication provides protection for the crew and meets the general requirements of safety. However, for the APOLLO mission involving lunar deorbit, this obviously is an unsatisfactory situation since propulsive impulse is essential to return the APOLLO capsule to earth. Thus, the dual responsibility of APOLLO propulsion is to provide reliability or to be fail-proof, in addition to being fail-safe.

It is perhaps this essential difference which indicates why engines for manned space rockets must meet far more stringent safety and reliability requirements than the rocket engines for manned aircraft. Requirements for aircraft rocket safety were set forth in MIL-E-5149, a general specification for aircraft rockets. This specification states that under any single condition of malfunction, or in certain cases of malfunction of powerplant supply, the rocket powerplant will shut down or react in a safe manner without creating a hazard to the aircraft. It follows that premature exhaustion or loss of propellants in an aircraft mission is equivalent to a safe rocket shutdown, and flight safety will continue to be maintained by gliding to earth or, in extreme emergencies, by returning the pilot by parachute.

However, in the case of spacecraft operations, the availability of propellants and a propulsion system to utilize them reliably to provide the necessary velocity increment and guidance corrections are generally mandatory. Thus, an inherently safe system requires:

1. Assurance that fuel is adequate prior to committing the vehicle to lunar orbit, and
2. Powerplant redundancy

The first of these requirements implies the use of an adequate fuel-energy management system discussed in the following section. The second implies a high reliability gained from redundant units and is discussed in Section 3.3.

3.2 FUEL ENERGY MANAGEMENT

Proper management of propellant reserves is an essential ingredient of any space mission. Space ventures, by their very nature, require finite velocity changes of varying sizes and durations depending on the actual mission requirement for midcourse correction, retro-propulsion, lunar orbiting, lunar orbit escape, etc. At the start of any mission, it is reasonable to assume that sufficient propellants will be available to provide the basic ΔV capability for the nominal mission plus a reserve allowance. If all goes well, the reserve will still be available when the mission has been completed. The fuel energy management system is required when the mission deviates from the nominal.

There are several conditions of malfunction resulting in greater propellant consumption than planned during midcourse and lunar orbiting maneuvers. Although a series of protective and diagnostic devices will be available on the APOLLO spacecraft, it is essential that a continual monitoring system be utilized to maintain a watch over remaining propellant reserves and to provide an intelligent course of action should unexpected loss of propellants occur. Such a system implies a device capable of sensing the mass of propellants remaining in the tanks—a device not presently available but which should be developed as rapidly as possible.

The fuel-energy management system will be available on-board to observe propulsion energy potentially remaining and to plot the strategy for a safe return to the earth considering numerous return paths and their related ΔV requirements. This fuel energy management system is described in considerably more detail in Appendix P-F.

3.3 ON-BOARD PROPULSION REDUNDANCY

An essential ingredient of reliability involves the use of redundant systems, subsystems, and components. Even for the simplest systems, powerplant redundancy can take several forms:

a. Common feed system - redundant chambers

b. Redundant feed system - common chambers

c. Redundant feed system - redundant chambers

If the reliability of individual elements is below an acceptable standard, the probability of multiple malfunctions becomes quite significant and increases the risk encountered by the flight crew. Therefore, to achieve safety resulting from high reliability, it is an absolute necessity to use redundant units of those individual elements which may have a lower-than-desired reliability.

During this study, Reaction Motors Division of Thiokol Corporation has provided some valuable insight into this question of redundancy, particularly with regard to the potential advantages of fail engines.

In the APOLLO program where safety and reliability are paramount, the case of engine redundancy can represent an important element in the achievement of both safety and reliability.*

Specifically, the selected 1963 main propulsion system incorporates four engines which afford redundancy in performing orbital change requirements, but result in weight and over-all performance penalties. If two engines are utilized with step thrust capabilities, it is possible to perform both orbital and abort requirements. The use of two engines would result in a weight saving and performance gain. However, the safety and reliability aspects of four engines versus two engines must be considered in detail to determine what penalty in safety one may expect to pay for the use of two engines.

There are two functions required of the main midcourse propulsion system, namely mission trajectory change and abort. Each function must be considered separately for the case of four engines and two engines.

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* Thiokol Chemical Corporation, Reaction Motors Division, Multiple Engine Trade-Off

3.3.1 Trajectory Change

3.3.1.1 FOUR ENGINES

To accomplish an orbital change, any one of the four engines is required to function. Therefore, the probability (R_o) of performing an orbit change can be expressed as follows:

$$R_o = 1 - (1 - R_1)^4$$

where R_1 is the reliability of a single engine.

Assuming a reliability of 90 percent for each engine for illustrative purposes only, the probability of performing the orbital change function is 99.99 percent. It is obvious that a quadruple malfunction is required before the orbital change function is destroyed.

3.3.1.2 TWO ENGINES

In this configuration, each engine is designed for step-thrust operation, that is, the engine is capable of 6,000 or 12,000 pounds thrust. This added requirement increases the engine complexity, thereby reducing reliability. For illustrative purposes only, it will be assumed that the engine reliability for the two-engine configuration is 89 percent. Therefore, the probability of performing the orbital changes is:

$$R_o = 1 - (1 - R_1)^2$$

or 98.79 percent. In comparison, the four-engine configuration offers greater safety and reliability in performing the orbit change function.

3.3.2 Emergency Escape (Abort)

3.3.2.1 FOUR ENGINES

The case of the abort function will depend directly upon the thrust level required or the number of engines required to function to achieve a safe abort. The mathematical expression for the four-engine configuration can be given as follows:

$$R_1^4 + 4R_1^3q + 6R_1^2q^2 + 4R_1q^3 + q^4 = 1$$

where,

R_1^4 = the probability of no malfunction or reliability of firing all four engines

$4R_1^3q$ = probability of a single malfunction

$6R_1^2q^2$ = probability of a double malfunction

$4R_1q^3$ = probability of a triple malfunction

q^4 = probability of a quadruple malfunction where $q = 1 - R$

Assuming a reliability of 90 percent for each engine for calculation purposes only, then, the following table applies:

Number of Malfunctions	Term	Thrust Level (lbs)	Probability of Occurrence (%)	Probability of Obtaining Specified Thrust or More (Cumulative Probability %)
0	R_1^4	24,000	65.61	65.61
1	$4R_1^3q$	18,000	29.16	94.77
2	$6R_1^2q^2$	12,000	4.86	99.63
3	$4R_1q^3$	6,000	.36	99.99
4	q^4	0	.0001	100.00

3.3.3.2 TWO ENGINES

The mathematical expression for the two-engine configuration can be given as follows:

$$R_1^2 + 2R_1q + q^2 = 1$$

where,

R_1^2 = The probability of no malfunctions or reliability of firing two engines

$2R_1q$ = probability of a single malfunction

q^2 = probability of a double malfunction with an assumed reliability of 89 percent for each engine, the following table applies:

Number of Malfunctions	Term	Thrust Level (lbs)	Occurrence (%)	Probability of Obtaining Specified Thrust or More (Cumulative Probability %)
0	R_1^2	24,000	79.21	79.21
1	$2R_1^2q$	12,000	19.58	98.79
2	q^2	0	1.21	100.00

From the above charts and Figure I-3-1 it can be seen that the level of thrust required to attain a successful abort will dictate which configuration would be more applicable to the abort function. If 24,000 lb of thrust were absolutely required, then the two-engine configuration represents a better approach. However, if malfunctions occur and a reduced thrust level is permissible, then the four-engine system represents greater safety and reliability than achieved with the two-engine configuration. From an over-all safety and reliability aspect, it is more realistic to choose the four-engine configuration. Safety cannot be compromised to achieve performance and weight gains. However, with increasing basic engine reliability (e.g., 0.95) the differences between the two- or four-engine systems become relatively insignificant which permits performance and weight consideration to play a larger role in determining the final approach. This illustrates the most important advantage of a four-engine system, namely, that the four engine system can be brought to a given required level of reliability with reduced development costs and in a shorter period of time. The major costs of providing reliable and safe systems lie in the development and demonstration programs, which can be minimized for early powerplant delivery. In the preceding example, we needed only to demonstrate an engine reliability of 89 percent to have an over-all propulsion system reliability of 99.99 percent. As higher engine reliabilities are demonstrated, the two-engine cluster would be acceptable, but would never quite attain the reliability of the four-engine cluster.

Possible interactions between engines have been briefly studied. Based on this cursory examination, no case of detrimental interactions having the most improbable multiple malfunctions have been discussed, and the analysis, as stated, seems valid. The four-engine cluster, therefore, has been retained for the proposed designs.

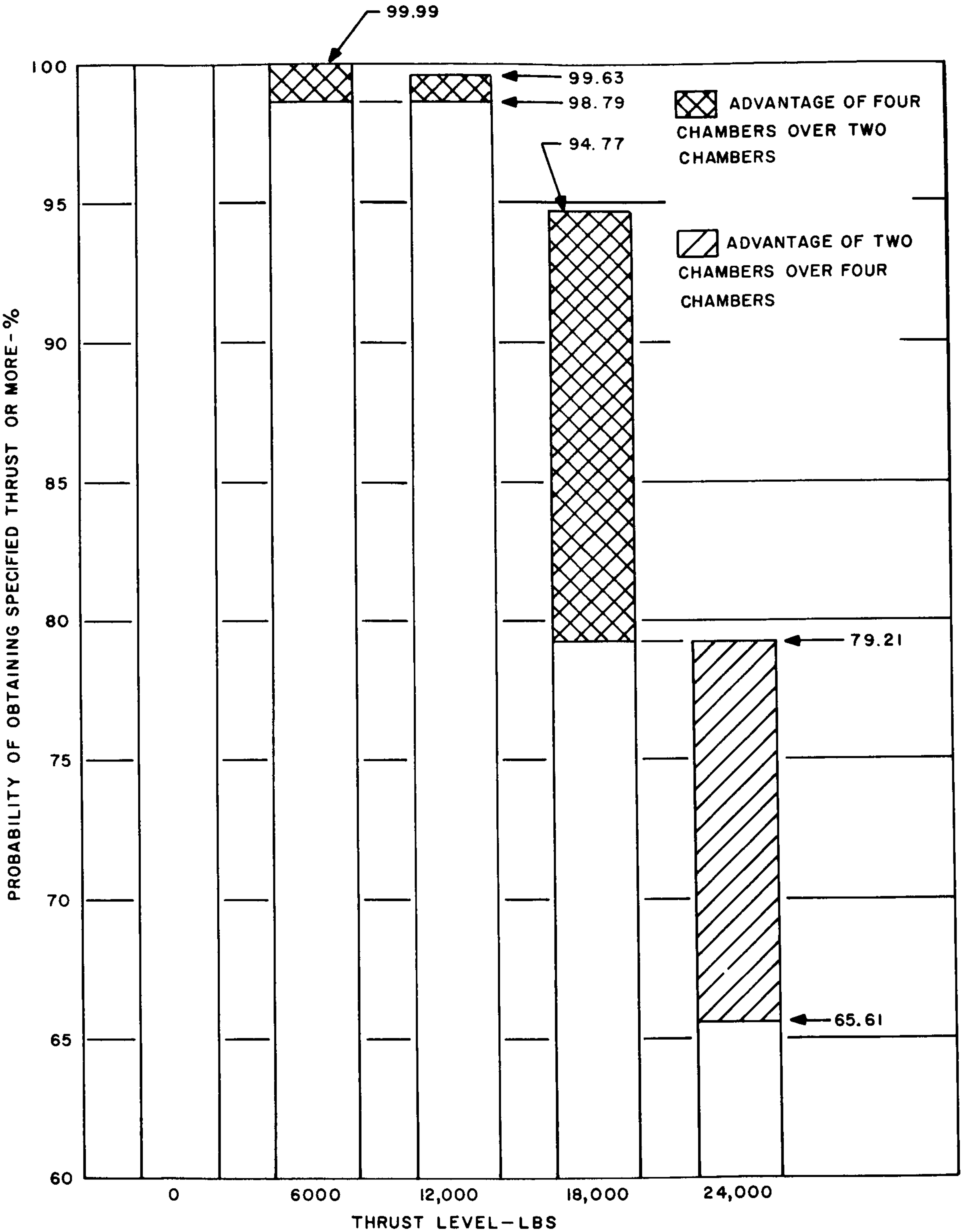

Figure I-3-1. Probability of obtaining specified thrust or more

3.4 ACHIEVEMENT OF SAFETY AND RELIABILITY IN MANNED SPACE VEHICLES

Reliability must be established as a design concept, built into the components, developed, demonstrated, and maintained throughout the development program. Safety is achieved through careful attention to every possible malfunction area with continued assurance that a combination of events cannot result in an unsafe condition. More specifically, safety is a natural dividend of a careful and thorough reliability program.

The reliability program may be logically divided into four phases:

1. Design (reliability synthesis)
2. Development program (reliability attainment)
3. Demonstration test program (reliability measurement)
4. Production and quality assurance (reliability maintenance)

During the design and development, reliability attainment is achieved through careful attention to both expected performance and safety. A series of analyses will be made to analyze the complete spectrum of any potential malfunctions throughout the flight period. These malfunction analyses then permit trade-off studies to be performed to relate potential improvements in reliability to mission parameters such as weight, cost, etc.

An example of reliability attainment is the use of redundant motors versus unit reliability. This reliability gain must then be weighed against the obvious losses of decreased payload weight, complexity, and cost.

For the advanced propulsion needs of APOLLO, it is mandatory that plans include ample consideration of reliability in the development program. With limited time and funds, the balance of work between design and development must be maintained to assure an adequate and safe propulsion system which will reflect the state-of-the-art at the time development testing has been completed.

Emphasis on adequate development of reliability rather than demonstration is the keynote here. It does little good to demonstrate existing or inappropriate components if the mission requires a higher performance obtainable with improved hardware. Obviously, development of reliable improved components must attend the achievement of the required higher performance to ensure an adequate, safe rocket system.

What then of reliability demonstration? The answer here is an intelligent balance between demonstrated reliability, confidence level, and program cost. The development and demonstration programs need to be designed to provide the best use of a modest budget, thereby ensuring the best over-all probability of mission success. For example, it might be advantageous to continue development of reliability of an abort rocket from, say 0.98 to 0.995, even though only enough reliability demonstration tests can be conducted to produce a confidence level of 50 per cent in the higher reliability. This is in contrast to alternate programs which might either spend the same amount of money running repeated tests of existing units to demonstrate the reliability of 0.98 with a confidence level of 99.5 percent; or an alternate design program to increase unit redundancy to give an overall 0.995 reliability but at the expense of increased weight and complexity.

Rigorous discipline and analysis are necessary here to ensure proper attainment of reliability, safety, and optimum utilization of propulsion technology.

The final area (3-4, above) of reliability maintenance remains immutably fixed as the most necessary adjunct to safe and reliable propulsion. None of the foregoing analyses can compensate in any manner for adequate production and quality control. Production of a reliable solid abort motor, for instance, can be achieved only with careful attention to each step of the fabrication. See appropriate quality control techniques such as grain x-ray, case Zyglo, Magnaflux, dimensional inspection, batch quality control, etc. are necessary for development, proof test and production of propulsion systems. When defects occur (for example, the separation of the solid charge bonding to the wall), it is possible to analyze these defects during development tests and develop an adequate knowledge of the nature of limitations of such components and a quality assurance code for final production units.

Such a program of continued reliability analysis, attainment, and maintenance, rather than strict adherence to any existing MIL specifications for manned rocket engines, should ultimately produce the most reliable and safe propulsion system for the manned APOLLO mission.

4.0 Main On-Board Propulsion

4.1 PARAMETRIC STUDIES

Two of the key elements in selecting a propulsion system design are the mission velocity requirements and the weight constraints. A lunar orbital mission was selected as the prime requirement for APOLLO. For this type of flight, a mission velocity increment of 7500 feet per second was utilized. This provides a total of 500 feet per second for midcourse correction on outbound and returning segments of the flight, 3500 feet per second entering the lunar orbit, and a similar value on leaving the lunar orbit. The derivation of the mission velocity increment is further discussed in Volume III, Trajectory Analysis and Guidance and Control.

For a vehicle available in the 1963 time period, the initial parametric analysis assumed that a total vehicle weight of 15, 715 lb could be utilized. With the input data outlined in Section 2.0, Introduction, the three propulsion subcontractors prepared power-plant designs. The results of these studies are given in the upper portion of Table I-4-I. Values are given for both the D-2 semi-ballistic re-entry vehicle and the R-3 modified lenticular re-entry vehicle throughout this section. Note that maximum payload is 6270 pounds for hydrogen-oxygen, 7003 pounds for hydrogen-fluorine, and 5541 pounds for the storable combination.

While the three propulsion system designs were in progress, design data became available for generation of the actual payload weights. These weights are shown at the top of the middle portion of Table I-4-I. Allowances for the propulsion module skin weight have been included in the payload to reflect the weight savings which accrue from the more compact designs of Bell and Thiokol. Further, data furnished by the Astronautics Corportion of America indicated that a 5 percent fuel reserve would be adequate as opposed to the 10 percent in the original specifications. Additional details on the fuel reserve may be found in Appendix P-F.

Combining these new data with the powerplant information, the total velocity capability and the required total vehicle weight was obtained for each configuration. Note

TABLE I-4-I. 1963 PARAMETRIC DESIGN COMPARISON DATA

	Company	Aerojet		Bell		Thiokol	
	Vehicle Type	D-2	R-3	D-2	R-3	D-2	R-3
Design Data per Specifications	Propellants	H_2/O_2	H_2/O_2	H_2/F_2	H_2/F_2	MMH/N_2O_4	MMH/N_2O_4
	Isp, Sec.	430	430	446	446	318	318
	Propulsion System, Fixed Wt.	1593	1593	1646	1646	1250	1250
	Propellant Wt. For Δ V = 7500 Fps, 10% Reserve + 3% Outage	7618	7618	6832	6832	8690	8690
	Payload for Δ V = 7500 Fps	$\boxed{6270}$	$\boxed{6270}$	$\boxed{7003}$	$\boxed{7003}$	$\boxed{5541}$	$\boxed{5541}$
	Weight at Booster Burnout	15,715	15,715	15,715	15,715	15,715	15,715
Design Data with Actual Payload	Actual Payload	7983	9025	7669	8711	7749	8791
	Revised Wt. at Booster Burnout	17,428	18,470	16,381	17,423	17,923	18,965
	Penalty Wt. (Abort Rockets, Adapter)	975	607	975	607	975	607
	Effective APOLLO Wt.	18,403	19,077	17,353	18,027	18,895	19,569
	Δ V Deficienty (Fig. I-4-1), Fps	1260	1550	900	1180	1500	1720
	Total Δ V Available with 5% Res., 3% Ullage	7220	6660	7040	6500	6380	5790
	Δ V Mission After Escape with 5% Res., 3% Ullage	$\boxed{5960}$	$\boxed{5110}$	$\boxed{6140}$	$\boxed{5320}$	$\boxed{4880}$	$\boxed{4070}$
	Δ V Mission After Escape No Res., No Ullage	6780	5850	6950	6070	5360	4600
Re-Des. for Same Escape Wt. as H_2/O_2	Δ V Mission After Escape with 5% Res., 3% Ullage	5960	5110	6630	5710	4680	3900
	Δ V Mission After Escape No Res., No Ullage	6780	5850	7580	6580	5490	4650

that the weight in each case exceeds 15,000 lb. With the presently available Saturn C-2 data, 15,000 lb is the maximum payload that can be boosted to escape velocity. However, if the APOLLO vehicle weight is greater than this value, the additional velocity required to achieve escape may be obtained by using the APOLLO propulsion. The velocity decrement is plotted as a function of stage weight in Figure I-4-1. To utilize this curve, 975 pounds must be added to the APOLLO stage weight at booster burnout to account for the adapter and the solid rocket penalty weight (See Section 5.0). The circled numbers in Table I-4-I indicate the mission velocity available after achieving escape velocity. The velocity is indicated for the case of complete propellant expenditure directly below this figure, that is, no reserves and no ullage. Note that only the D-2 configuration with either F_2/H_2 or O_2/H_2 will achieve 5600 feet per second or more for super orbital abort with allowance for reserves and ullage. The resulting velocities for the MMH/N_2O_4 combination are well below 5600 even with 100- per cent propellant expenditure. The R-3 vehicle results in approximately a 1000-ft/sec velocity degradation compared to the D-2 vehicle for the two cryogenic designs.

Finally, to compare the designs on an equivalent basis, the total vehicle weights (17,428 lb for the D-2 and 18,470 for the R-3) for the oxygen-hydrogen combinations were used. The resulting velocities are shown in the bottom section of Table I-4-I. These data indicate that the hydrogen-fluorine combination increases the velocity capability in the order of 700 ft/sec as compared to the hydrogen-oxygen system. Comparative data for the three propellants are shown in Figure I-4-2.

The next facet of this study concerned itself with a parametric analysis of the 1966 systems. Potential powerplant improvements for the hydrogen-oxygen and hydrogen-fluorine systems have been factored into these analyses. In addition, Thiokol's Reaction Motors Division has incorporated oxygen difluoride (OF_2) into their system to replace nitrogen tetroxide. The fuel is monomethyl hydrazine as before. Utilizing Figure I-4-1 and parametric data supplied by the subcontractors, the data in Figure I-4-3 were compiled. These curves show the relationship between vehicle weight and payload for the designs under consideration. As discussed in Volume VIII, Preliminary Design, it may be possible to reduce the D-2 vehicle weight to approximately 7000 pounds in the 1966 time period. Referring to Figure I-4-3, note that the

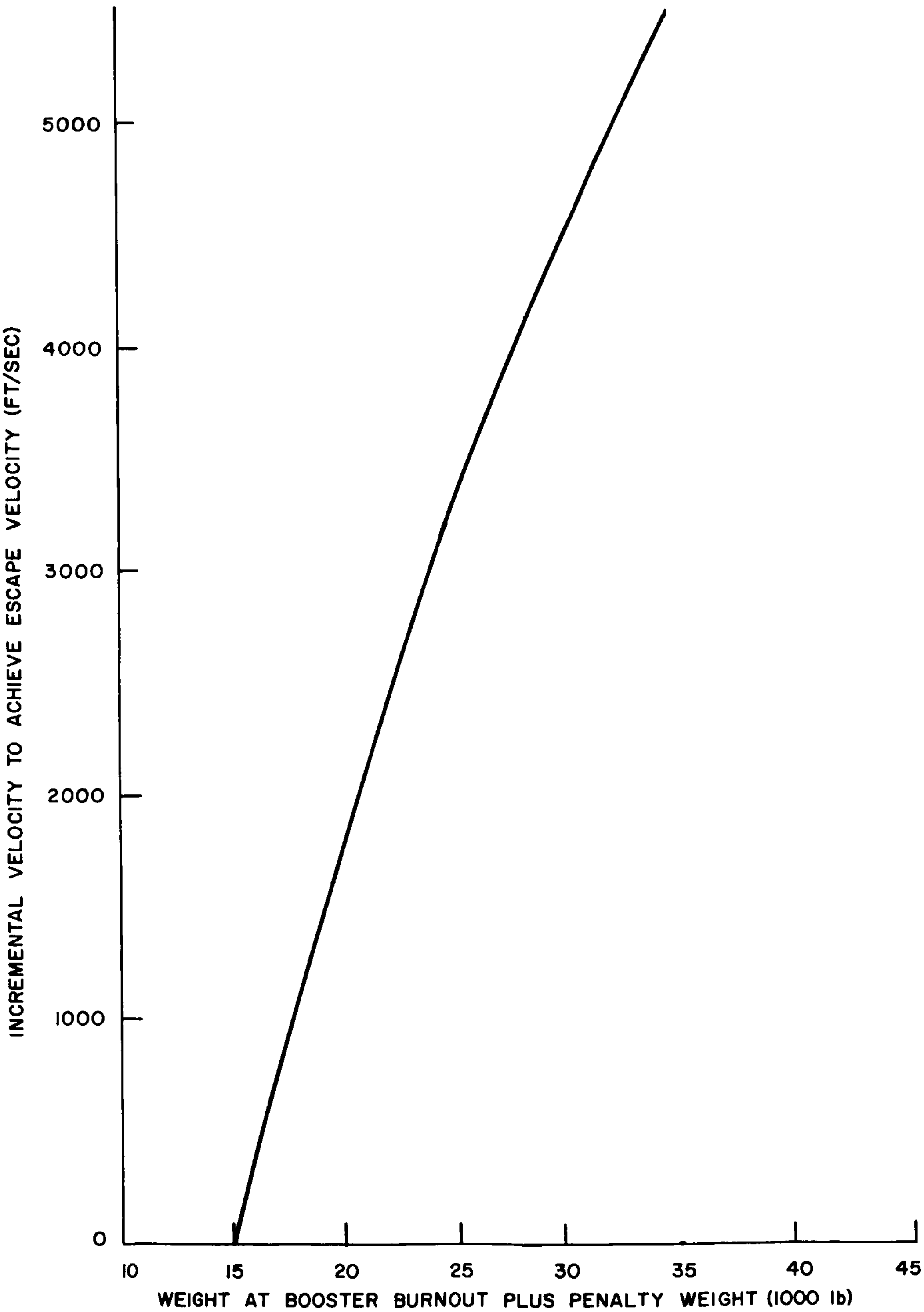

Figure I-4-1. Saturn C-2 velocity deficiency

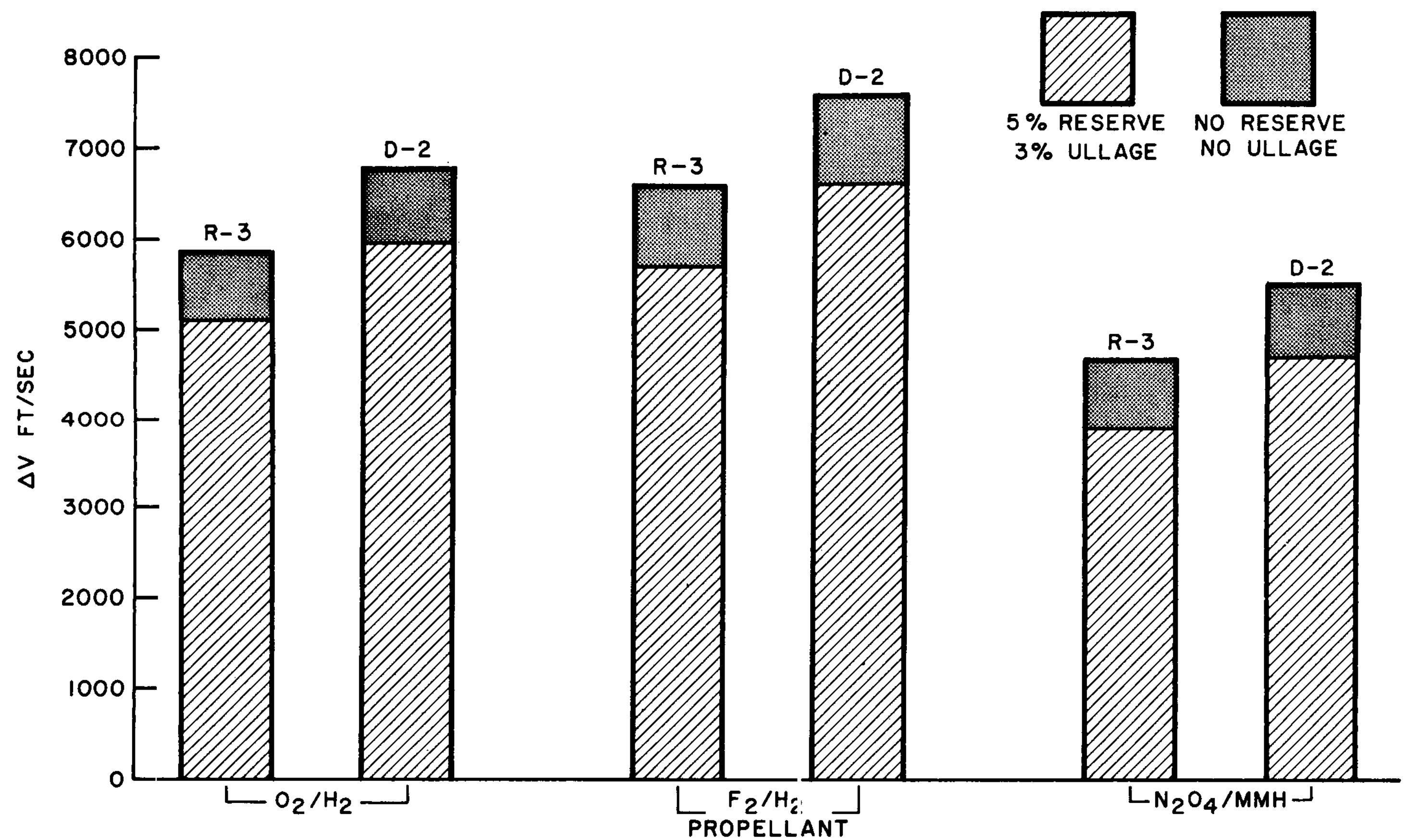

Figure I-4-2. Comparative propellant data, 1963 systems, vehicle weight = 17,428 pounds

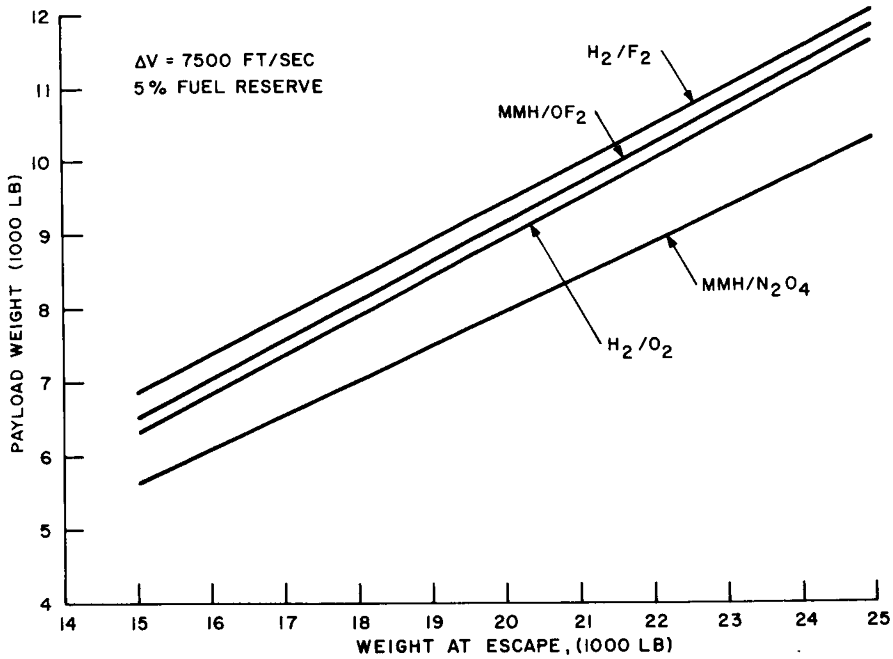

Figure I-4-3. Payload vs weight at boost burnout

vehicle weight will still be in excess of 15,000 lb for any of the systems shown. Thus, unless the performance of the Saturn C-2 vehicle exceeds the present estimates, it will be necessary to modify the mission. Among the possible alternatives are:

1. Reduction in payload by decreasing the number of crew members
2. Lowering the velocity requirements by eliminating the lunar-orbit mission
3. Decrease safety aspects by lowering of eliminating fuel reserves and eliminating redundancy

A preferable alternative which would allow the basic mission to be accomplished without resorting to any of the above compromises is to utilize the APOLLO main on-board propulsion to make-up the velocity deficiency accruing from stage weights in excess of 15,000 pounds. The results of a parametric analysis for this concept are summarized in Figure I-4-4. In this concept, payload weights of over 8000 lb may be utilized without excessive APOLLO stage weights. In Table I-4-II data are presented for payloads of 7000, 8000, and 9000 lb as taken from Figure I-4-4. This table shows that there is only a 2700-lb increase in the total weight of an O_2/H_2 system compared to an F_2/H_2 system for an 8000 pound payload. Since both weights

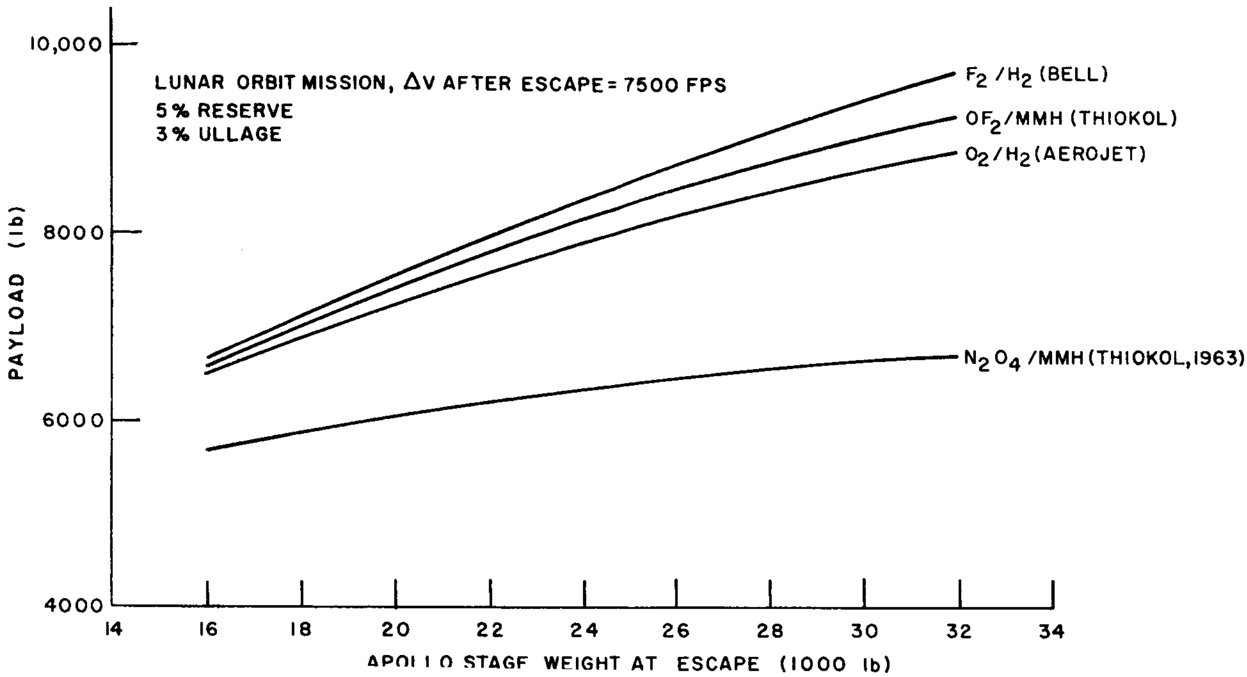

Figure I-4-4. APOLLO payload vs. stage weight 1966 design

are considerably in excess of 15,000 lb, the utilization of fluorine with its attendant handling problems does not appear to be warranted. If a heavier vehicle weight should prevail (such as is typified by the 1963 weight of the R-3 glide vehicle), the fluorine system may then be necessary. Although the OF_2/MMH combination proposed by Thiokol appears to offer a payload advantage as compared to oxygen-hydrogen, the lack of test data would involve too great a risk in depending upon its use for the APOLLO mission.

TABLE I-4-II. 1966 VEHICLE WEIGHTS

Payload, Lb	APOLLO Stage Weight, Lb		
	O_2/H_2	F_2/H_2	OF_2MMH
7000	18,800	17,600	18,000
8000	25,000	22,300	23,300
9000	36,000	27,600	30,100

The use of solid propellants for the APOLLO mission has also been studied. The best performance of solids is essentially equivalent to that of N_2O_4/MMH which is insufficient for this mission. If ΔV requirements were lowered, solids would be competitive with the high-energy cryogenic systems. Additional data on a solid propellant design are given in Appendix P-G.

Additional parametric studies have been conducted to compare pumped and pressurized systems. Data provided by the Aerojet-General Corporation are shown in Table I-4-III and are graphically illustrated in Figure I-4-5. Two types of pressure-fed systems are considered. In the Hylas system, heated helium is used to pressurize the hydrogen tank. Propellant transfer in the VaPak system is accomplished by maintaining the propellants at or near the temperature corresponding to the saturation pressure required. Opening the propellant valves then drops the pressure and permits surface boiling to generate the pressurizing gases. Some pressure (and hence thrust) decay is inherent in this system.

TABLE I-4-III. COMPARISON OF PRESSURIZED AND PUMP-FED SYSTEMS DATA PROVIDED BY AEROJET-GENERAL CORPORATION

ASSUMPTIONS:

Initial Weight = 16,000 lb

ΔV = 7500 ft/sec

10 percent Fuel Reserve,
3 percent Ullage

Feed System	Pressure Fed			Pump Fed	
Propellants	O_2/H_2 (Hylas)	O_2/H_2 (VaPak)	F_2/H_2	O_2/H_2	F_2/H_2
PROPELLANT					
Usable	6690	6690	6480	6745	6550
Reserve and Outage	870	870	890	875	850
Boil-off	0	0	0	5	90

TABLE I-4-III. COMPARISON OF PRESSURIZED AND PUMP-FED SYSTEMS DATA PROVIDED BY AEROJET-GENERAL CORPORATION (Cont)

Feed System	Pressure Fed			Pump Fed	
Propellants	O_2/H_2 (Hylas)	O_2/H_2 (VaPak)	F_2/H_2	O_2/H_2	F_2/H_2
TANKAGE					
Fuel Tank & Insulation	291	351	129	237	129
Oxygen Tank & Insulation	111	126	97	112	99
PRESSURIZATION	248	201	115	41	28
STRUCTURE	227	227	220	232	222
ENGINES	463	463	463	620	490
ATTITUDE CONTROL	267	267	267	267	267
TOTAL PROPULSION SYSTEM	9167	9195	8611	9234	8725
PAYLOAD	6833	6805	7389	6766	7275

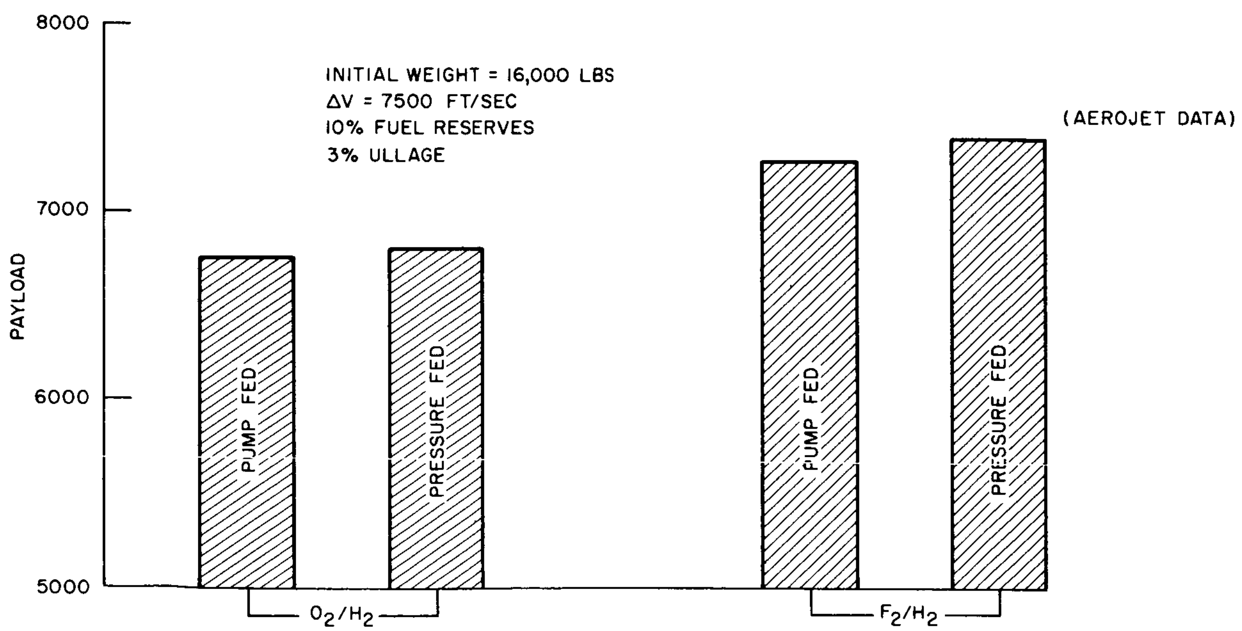

Figure I-4-5. Comparison of pressurized and pump fed systems

The pressure-fed systems shown in Table I-4-III are assumed to consist of a cluster of four 6000-lb thrust ablatively cooled thrust chambers. These units operate at a thrust chamber pressure of 65 psia and deliver a vacuum specific impulse of 430 seconds. The pump-fed systems consist of a two-thrust-chamber configuration using two 17, 500-lb thrust pump-fed engines delivering 426 second specific impulse and estimated to weigh 310-lb each. Pump suction head for these engines is obtained from a booster pump included in this weight. Pressurization system weight is then only the saturated propellant vapor remaining in the tank after expulsion of the liquid. If a separate pressurization system is required for pump suction head, additional weight and complexity would be involved.

Examination of Table I-4-III and Figure I-4-5 shows that the performance advantage normally associated with the pump-fed system can be greatly reduced or even negated when compared to a pressure-fed system for extended vacuum usage. This is due to the feasibility of operating the pressure-fed systems at low (40 to 65 psia) pressure. Propellant tank thicknesses are at or near minimum gage, and there is virtually no loss in specific impulse at low chamber pressure for vacuum operation. Further, this analysis does not include the propellant lost during start cooldown which, for the Centaur engine, may be 60 pounds per engine per start. With adequate allowance for this loss, the pressurized system is obviously superior to the pumped system.

A comparison of pumped and pressurized systems was also conducted by the Bell Aerosystems Company for the hydrogen-fluorine combination. Their results indicated that a completely pressurized loaded propulsion system would weigh approximately 7 percent more than their combination pumped/pressurized design. Bell's analysis shows that this pressurized system dry weight is nearly 25 percent heavier than their actual design proposed. For their pumped/pressurized system, Bell has utilized two 12, 000-lb thrust chambers, while their pressurized analysis considered four 6000-lb thrust chambers. A breakdown of the factors contributing to the weight increase indicated 200 lb of additional propellant attributable to the reduced expansion ratio of the pressurized units, and a small performance degradation when operating at a chamber pressure of 50 psia. Bell's cold helium pressurization

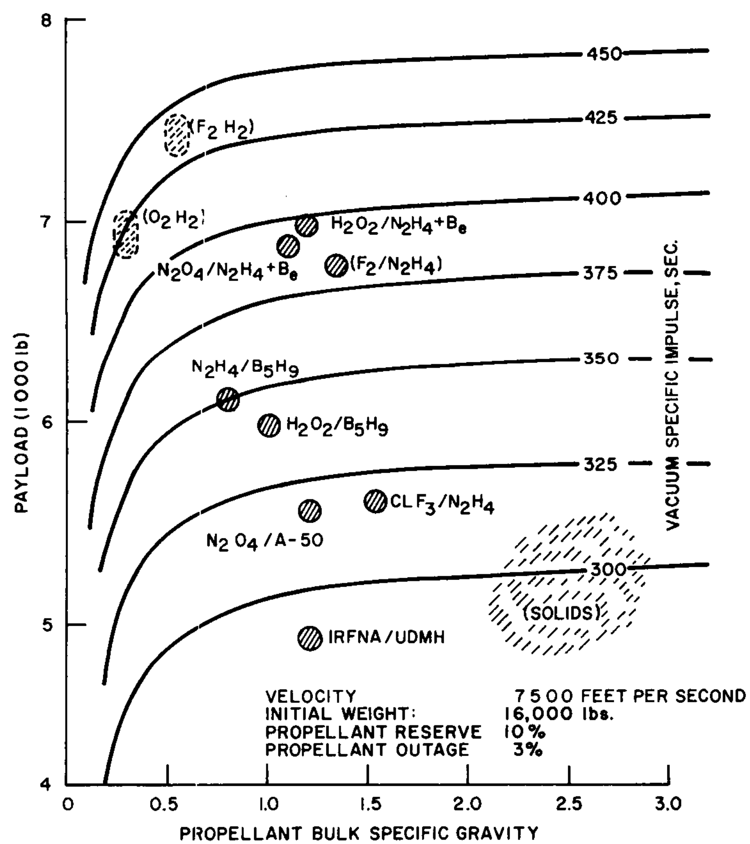

Figure I-4-6. Payload capability of various propellant combinations (AEROJET-general data)

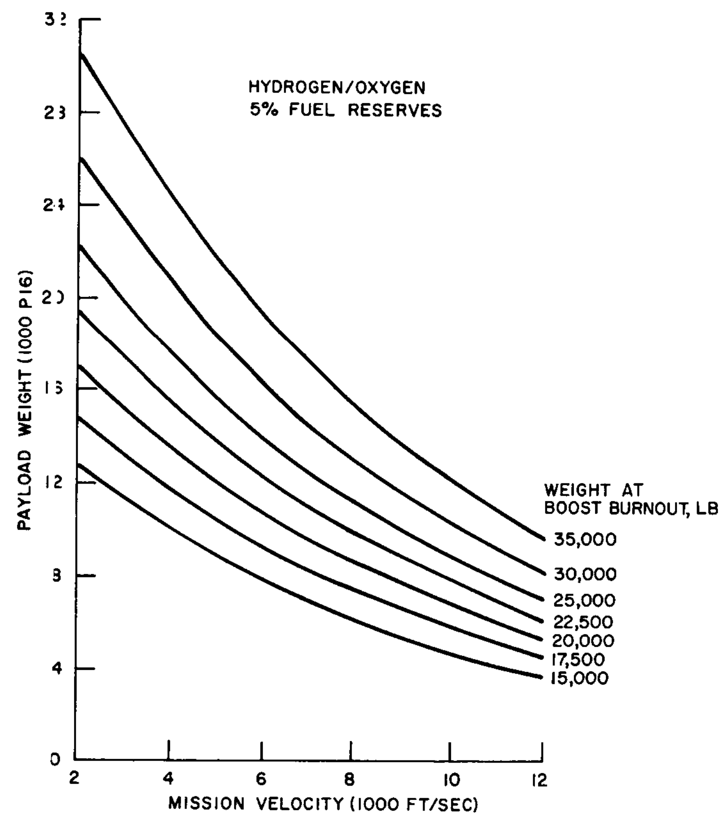

Figure I-4-7. Payload weight vs mission velocity requires an additional 200 lb for the pressurization system. A further source of weight increase, 100 lb, is attributable to the extra thrust chambers.

Figure I-4-6 shows a summary of the range of payload capabilities of various propellant combinations as calculated by the Aerojet-General Corporation. The figure is based upon a mission velocity of 7500 ft/sec and assumes a pressure-fed system operating at 100 psia chamber pressure with an expansion ratio of 40:1. It indicates that payloads in excess of 7000 lb can be obtained only with the high-energy cryogenic combinations or with the more advanced storable systems using light metal hydrides or slurries. Sufficient experience with these latter propellants does not exist to recommend them for use on manned vehicles in the time span under consideration.

In Figure I-4-7, payload as a function of mission velocity is shown for the hydrogen/oxygen propellant combination. Values are given for several weights at boost burnout.

On the basis of the parametric studies outlined here, it is quite evident that the total requirements of the APOLLO mission can be best satisfied by the application of a pressurized hydrogen-oxygen propulsion system.

4.2 SYSTEM SELECTED

4.2.1 Key Features

The on-board propulsion system for APOLLO has been selected to meet the basic mission requirements of safety, reliability, and performance. From the parametric and design studies, it appears evident that the on-board propulsion system can meet these objectives and provide both instant abort impulse for super-orbital return as well as the necessary velocity increment for lunar orbit and return. Use of a pressure-fed liquid hydrogen/liquid oxygen rocket engine provides the requisite high performance, yet permits attainment of the design objectives of reliability and safety.

The Aerojet-General on-board propulsion system, AJ-10-133, satisfactorily meets the requirements for the APOLLO mission and has been selected for the recommended vehicle. Other propulsion systems have significant advantages in specific areas, and are discussed in subsequent sections of this report.

Key features of the selected Aerojet-General AJ-10-133 Propulsion System are tabulated in Table I-4-IV.

TABLE I-4-IV. KEY FEATURES OF APOLLO MAIN PROPULSION SYSTEM (AEROJET-GENERAL AJ-10-133)

a. Designed specifically for manned space flight
b. High performance (I_{sp} = 430 sec) at low chamber pressure (65 psia)
c. Safe, Reliable
d. Versatile, potential growth
e. Simple, reliable, pressure-fed propellants
f. Single powerplant for all maneuvers
g. New, super-insulation (SI-4) permits sealed storage for fourteen days
h. Simple, proven ablative thrust chambers
i. Redundant thrust chambers and critical components
j. Proven pressurization system(s)
k. Reliable ignition (4 igniters per chamber + O_3F_2 for hypergolicity)
l. H_2/O_2 propellants are safe, nondetonable, nontoxic, noncorrosive, readily available to the engine
m. Compatible with space environment
n. Fuel energy management system
o. Instant readiness for super-orbital abort (24,000 lb thrust)

These key features are discussed below.

a. The proposed APOLLO engine is designed specifically for manned space flight and incorporates existing technology and components where applicable. The propulsion system can thereby be built up as an integrated system to meet the vehicle requirements of safety, reliability, and performance rather than attempting to compromise the APOLLO to existing engines which are neither designed nor qualified for manned space flight.

b. The engine will provide the requisite high performance (I_{sp} = 430 sec) with liquid H_2/ liquid O_2 at a low chamber pressure of 65 psia which facilitates safety and reliability.

c. Using representative numbers for unit reliability (see Paragraph 4.2.2), the probability of providing safe propulsion throughout the mission should be at least 0.978 and the probability of achieving a successful mission of approximately 0.954.

d. The single-propulsion package has both versatility and progressive growth capacity. The basic vehicle for 1963 weighs 18,000 lb at booster burnout for a payload of 7940 lb, and if boosted to lunar trajectory velocity, provides for a velocity increment of 7500 ft/sec, or provides sufficient propulsion to carry itself to escape and at a velocity of at least 6000 ft/sec after escape. Undertanking the propellants permits reduction of the vehicle weight to 14,025 lb which can be carried to escape by the Saturn C-2 or orbited by the Saturn C-1 with sufficient velocity for super-orbital abort. Thus, the complete powerplant can be checked out early in the program under actual operating conditions and the propulsion impulse increased for later lunar flights. Growth of the powerplant can be readily achieved, so that by 1966 the vehicle should be capable of achieving the lunar orbit mission.

e. The APOLLO propulsion features the simplicity and reliability of a pressurized-fed system. Such a system is inherently simple, should be available at an early date at low cost but with high performance. Pump-fed systems have been compared in many configurations but cannot better the payload-carrying capacity achievable with H_2/O_2 at 65 psia in a vacuum. Further, pumped engines are complex and require considerable conditioning for proper engine starts. This means that storage of LH_2/LO_2 would be difficult and inefficient with a significant weight of propellants lost in cooling down the engine to reach temperature equilibrium during starts. In addition, throttling to reduced thrust on a single chamber is quite feasible.

f. The single powerplant is capable of providing all necessary velocity increments during the APOLLO mission. This includes midcourse corrections, lunar orbit and de-orbit, and any other required maneuvers.

g. Use of the new super-insulation, such as SI-4, readily permits storage of liquid H_2/ liquid O_2 for the fourteen-day mission without excessive pressure use or the need for venting a most difficult task in a zero-gravity trajectory.

h. The simplicity and proven reliability of the ablative thrust chambers should greatly enhance the over-all system reliability. At the low chamber pressures of 65 psia, it is easily within the state-of-the-art for any single chamber to operate for the entire burning time of 546 seconds. Without cooling passages, start and shutdown can be quite rapid, (about one second) with minimum loss of H_2. Further, these chambers avoid potential leakage areas associated with regeneratively cooled chambers. With the ablative cooled chamber walls, operation of a single chamber at partial thrust is facilitated, and starts can be made at low flow, if desired, to settle the propellants.

i. Multiple redundant thrust chambers, tanks, and critical valves ensure high reliability and safety. For illustrative purposes, if chambers have a demonstrated reliability as low as 89 percent, two units raise the reliability to 98.74 percent and four units raise reliability to 99.99 percent. This in turn, should lower the cost and speed development, since for this illustration it is only necessary to demonstrate a chamber reliability of 89 percent. This can be done with only a few engine tests. As chamber reliability rises with development, it would probably be advantageous to consider the use of two chambers with double the thrust.

j. The proposed engine utilizes the proven pressurization system developed by Aerojet-General for the Hylas engine under AF Contract 04(611)-5170. An alternate pressurization system (which is probably mutually compatible with the proposed Hylas system) is designated VaPak by Aerojet and should provide a "belt and suspenders" redundancy for feeding propellants. The initial pressurization system, out to lunar orbit, will be the Hylas type, pressurizing the liquid O_2 with heated H and the liquid H_2 with heated gaseous H_2. The return from the Moon could be achieved with either the Hylas or the VaPak system — either of which should be adequate for pressure feed. Thus, if the "belt" fails, the "suspenders" can still keep the pants up.

k. Demonstrated reliable ignition of the H_2/O_2 using four surface-gap spark plugs in each of the chambers should further enhance reliability. Further ignition safety may be incorporated by using O_3F_2 which has been shown to produce hypergolicity of these propellants by Temple Research Institute. As little as 0.05 percent O_3F_2 has reliably produced ignition when in solution with the liquid O_2 in small thrust chambers. More research is needed on O_3F_2.

l. The propellants selected (O_2 and H_2) are safe, nonexplosive, nontoxic, noncorrosive, and are readily available. Excellent experience is available from over a decade of testing, handling, and storage. The propellants are compatible logistically with the upper stages of Saturn, and are daily being handled safely on a tonnage basis.

m. The proposed system is compatible with space environment. The natural vacuum of space facilitates storage and permits operation of the thrust chamber at high performance with low chamber pressure. Protection and redundancy of components provide safety in the space environment of radiation and meteorites. The ablative chambers and radiation cooled skirts are fairly resistant or unsensitive to meteorite puncture.

n. A fuel energy management system is provided for conservation and best utilization of remaining propellants, particularly in the event of a malfunction. Further, there is the possibility of manually monitoring the utilization of propellants to assure minimum residual propellants.

o. The pressurized system will be in readiness during boost so that superorbital orbits can be effected with rapid (one second) application of full, 24,000-lb thrust.

Other features of the selected system are described in the Aerojet report, Appendix P-A. Specific examples of other possible advantages include use of the heated H_2 alone for attitude control (I_{sp} of H_2 gas is 200 seconds at 270 degrees R); use of the O_2 for breathing in an emergency; use of the H_2/O_2 for the fuel cells in an emergency; and possible use of the settling jet for small corrections in ΔV or for precise impulse termination.

4.2.2 Main Propulsion System Design

4.2.2.1 GENERAL

The basic propulsion system selected for discussion here is the Aerojet-General AJ10-133 system described in the Model Specification in Appendix P-A, Aerojet's section. This engine, shown in Figure I-4-8, is designed to be available for flight in 1963, and may be used with either the D-2 direct re-entry vehicle or the R-3 lenticular vehicle. Gross weight at boost termination, if propellants are completely loaded, is 18,000 lbs for the D-2 vehicle.

Performance with these two vehicles is shown in Table I-4-V. In each case, the total weight exceeds the allowable weight of 15,000 lbs which the Saturn can boost to escape. For purposes of this discussion, the analysis will be confined to performance of this powerplant with the D-2 vehicle, although obviously the same reasoning would apply to the R-3.

4.2.2.2 D-2 PROPULSION PERFORMANCE AND WEIGHT

The actual weight and performance with the D-2 are shown on Table I-4-VI. For a vehicle weight of 18,000 lb at boost termination, a payload of 7940 lb may be given a velocity increment of 8450 ft/sec. Part of this propulsion (1440 ft/sec) can be used for escape, leaving a capability of over 6000 ft/sec for maneuvering after escape or super-orbital abort. Or, the powerplant is capable of giving the stage a velocity of 7500 ft/sec with 5 percent reserve, 3 percent outage.

During the 1963 period, the basic AJ10-133 powerplant will be available for earth-orbital and near-space missions. The propellant may be undertanked as illustrated in Table I-4-VI to provide the basic 15,000 lb which the Saturn can boost to escape, thus reducing the total available velocity to 4840 ft/sec. A combination of reducing

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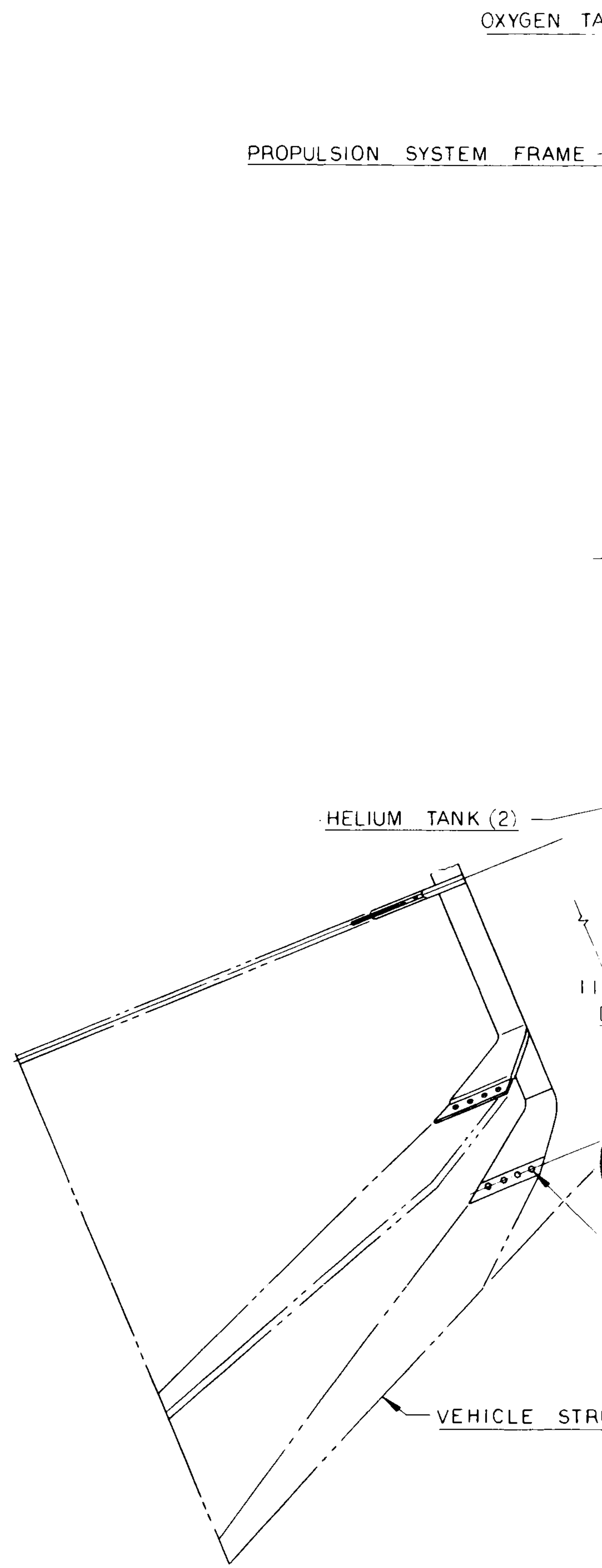
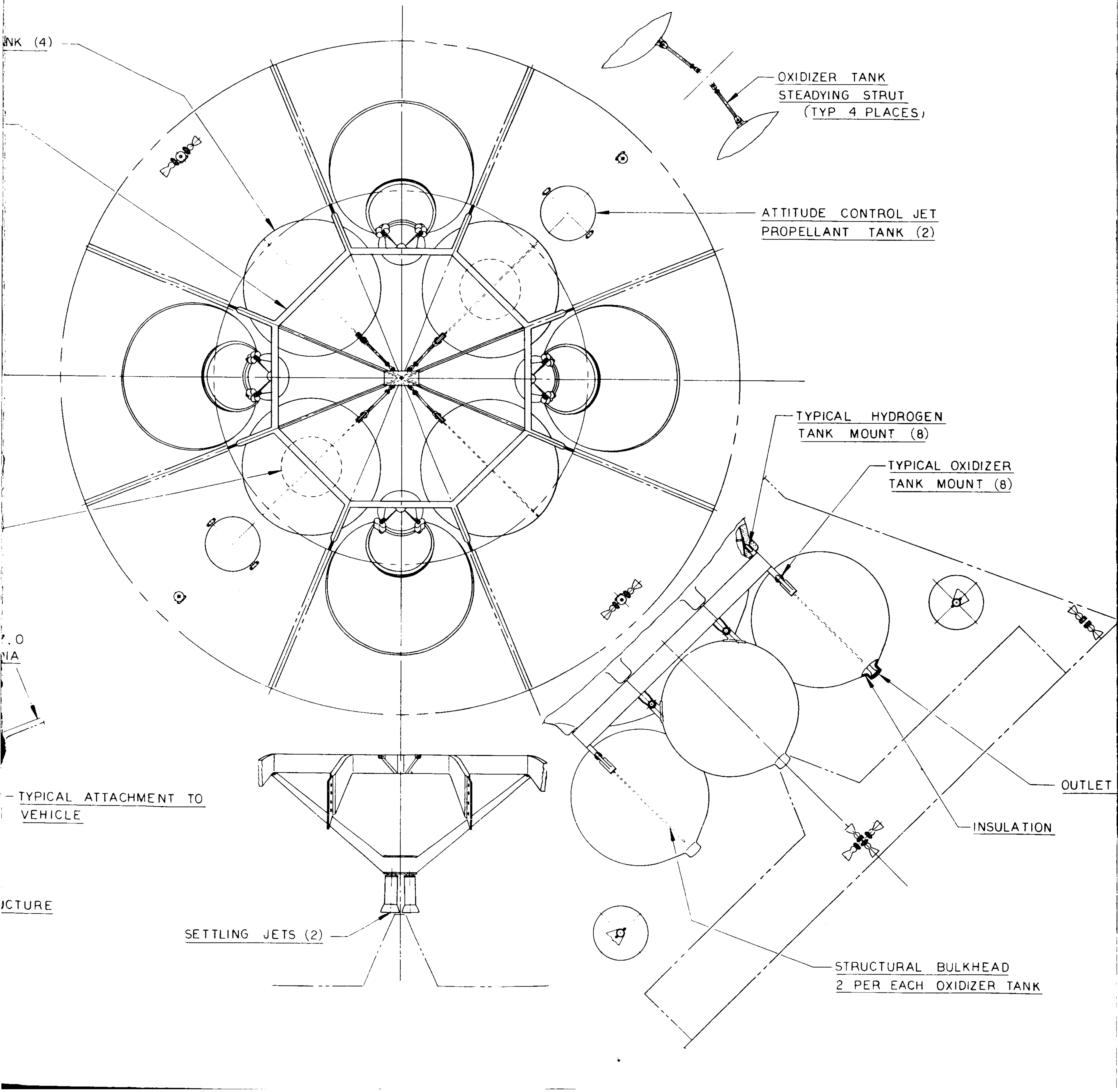
Note: In discussing the AJ10-133 APOLLO powerplant, it has appeared appropriate to restate much of the material prepared by the Aerojet-General Corporation. The attempt has been made to bring this material into sharper focus, but this has necessitated repeating some of their material. Specific credit is given where possible, and reference is made to Appendix P-A of this report for more complete details of the AJ10-133 propulsion system.



OXYGEN TA
PROPULSION SYSTEM FRAME
HELIUM TANK (2)
VEHICLE STR



NK (4)
OXIDIZER TANK STEADYING STRUT (TYP 4 PLACES)
ATTITUDE CONTROL JET PROPELLANT TANK (2)
TYPICAL HYDROGEN TANK MOUNT (8)
TYPICAL OXIDIZER TANK MOUNT (8)
OUTLET
INSULATION
STRUCTURAL BULKHEAD 2 PER EACH OXIDIZER TANK
SETTLING JETS (2)
TYPICAL ATTACHMENT TO VEHICLE
ICTURE
.0
IA

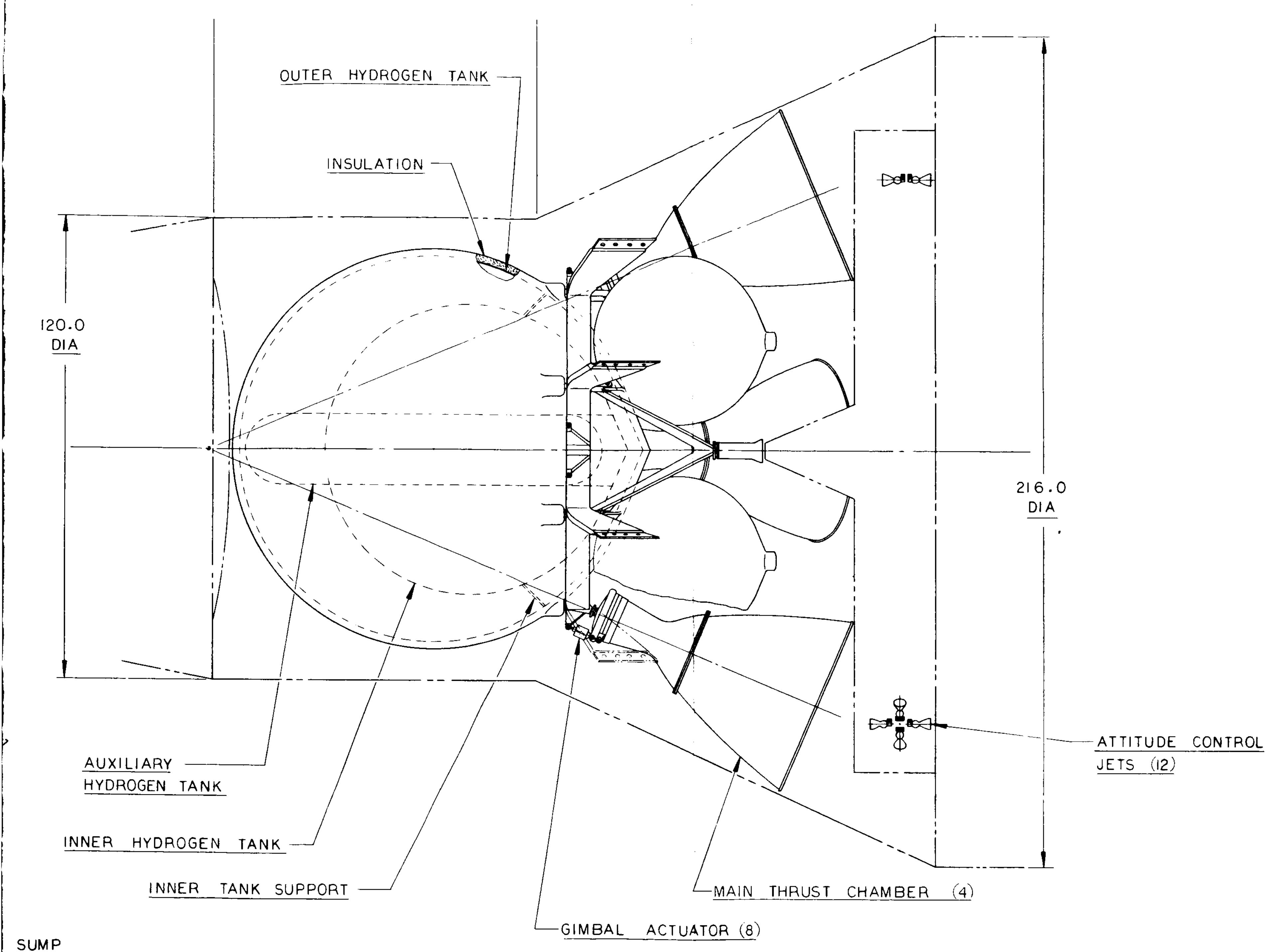


Figure I-4-8. Aerojet-General AJ10-133
APOLLO engine layout

[redacted]

TABLE I-4-V. LIQUID OXYGEN/LIQUID HYDROGEN 5 PERCENT RESERVE

	D-2 Re-entry Vehicle	R-3 Lenticular Vehicle
PENALTY WEIGHTS, lb		
Adapter	326	200
Small Separation Rockets	43	---
Large Separation Rockets	54 (335)[1]	---
Abort Rockets	552 (1829)[1]	407 (1128)[1]
Total Penalty Weight	975	607
Payload Weight	7940	9025
Propulsion System Weight (5% Reserve Propellants)	10,060	10,060
Useful Weight at Boost Termination	18,000	19,085
Total Weight on Pad	20,520	20,413
Total ΔV 5% Reserve (Stage Velocity) ft/sec	7500	7000
Total ΔV 3% Ullage No Reserves, ft/sec No Ullage	8450	7660
Mission ΔV (After Escape) ft/sec (5% Reserves, 3% Ullage)	6060	5200
Mission ΔV (After Escape) ft/sec (No Reserves, No Ullage)	7010	5860

[1] Total on the Pad Weight

TABLE I-4-VI. SUMMARY OF APOLLO D-2 PROPULSION WEIGHTS AND PERFORMANCE

	Weight, lb			
	1963 System Under-tanked	D - 2 Vehicle Basic Proposed System	1966 System D - 2A Vehicle	D - 2X Vehicle
Vehicle Weight @ Boost Burnout	14,025	18,000	19,300	25,600
Total Vehicle Weight on Pad	16,545	20,520	21,833	28,133
Payload Weight	7940	7940	7000	7983
Propulsion System Weight (Incl. att Contr., sep. rockets)	6085	10,060	12,417	17,734
Propulsion Fixed Weight (Incl. gas, att. cont. units)	1684	1684	1725	1900
Burnout Weight (No Reserve or Ullage)	9858	9741	8842	10,000
Available Propellants Weight*	4167 (tanks not full)	8142 (tanks filled)	10,458	15,600
	Δ V, ft/sec			
Maximum Δ V** (assuming use of reserves)	4840	8450	10,820	13,600
Δ V Used to achieve escape	None	1440	2050	3740
Δ V after escape w/5% res., 3% outage	3820	6060	7500	7500
Δ V after escape, no reserve	4840	7010	8770	9860
Δ V of stage with 5% res., 3% outage		7500	9550	11,240

* Does not include attitude control propellants

** Calculation for H_2/O_2, I_{sp} = 430 sec the payload (from 7983 to 7384 lb) and undertanking provides a capsule in the 15,000-lb class which has the capability of 5600-ft/sec velocity for super-orbital abort.

Thus, the basic powerplant can be proven, along with the APOLLO capsule in numerous missions prior to cis-lunar flights. The curve in Figure I-4-9 illustrates the range of ΔV achievable with this powerplant by undertanking the propellants. For example, the complete APOLLO vehicle could be launched with the Titan II vehicle at a weight of approximately 12,000 lb and provide a ΔV of 2600 ft/sec to help get the APOLLO capsule into a low earth-orbit and de-orbit. This would permit an early test of the capsule and propulsion system.

Improvements and weight reductions, available during this period, should permit reduction of the payload and powerplant specific weights so that by 1966 the D-2A vehicle should be realizable. This vehicle is illustrated in column 3 of Table I-4-VI and would have a vehicle weight at boost burnout of 19,300 lb for a payload weight of 7000 lb. This vehicle would then be capable (in 1966 when the C-2 booster was available) of propelling itself out to the Moon, orbiting and de-orbiting, and returning to the earth.

The D-2X vehicle represents a backup for the consideration of how the 1963 payload of 7983 lb could be orbited around the Moon and returned. Here, with today's state-of-the-art, this mission can be accomplished, but with a vehicle weight of 25,600 lb at boost completion.

4.2.2.3 ENGINE DESCRIPTION

The propulsion package for the D-2 configuration will utilize existing technology and components, where suitable, to provide a simple, reliable, high-performance rocket engine system. Selection of a pressurized propellant-fed system facilitates achievement of these goals by means of simple, uncooled, ablative thrust chambers similar to those developed by Aerojet General under Contract AF 04(611)-5170.

The configuration selected was determined by the thrust level required and envelope requirements. Super-orbital abort maneuvering necessitates a thrust of 24,000 lb for an average acceleration of about 2g's.

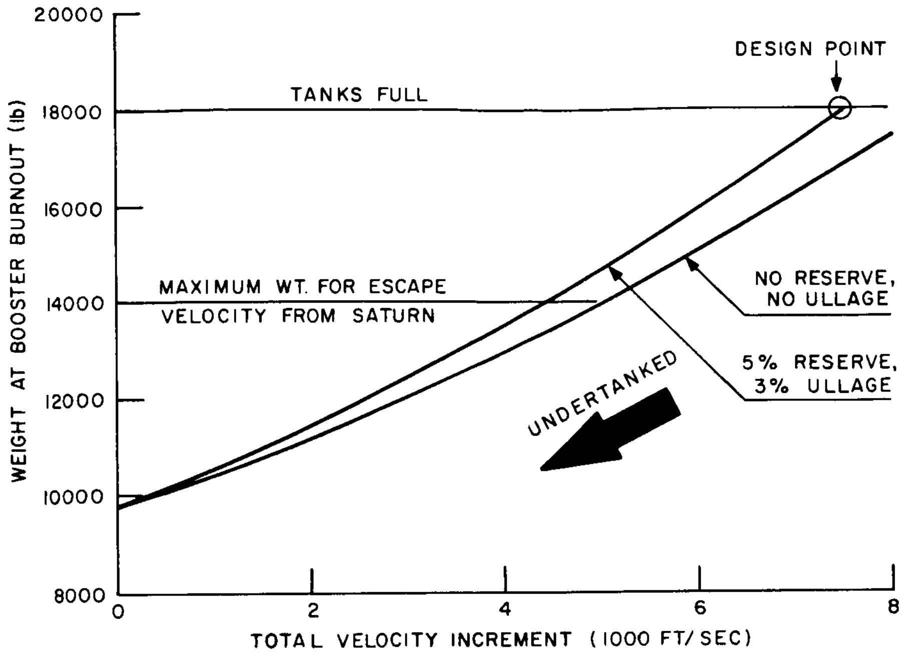


Figure I-4-9. D-2 vehicle performance payload = 7940 lb

Four individual chambers of 6000-lb thrust were selected, each capable of providing the necessary space maneuvers alone. All four chambers are fired for 24,000 lb thrust required in super-orbital abort. Two chambers of 12,000-lb thrust, throttleable to 6000 lb, would fulfill the same mission but exceed the available length. Therefore, the proposed engine is designed around the four chambers which provide an excellent reliability with high redundancy.

A summary of the 1963 engine dry weights is shown in Table I-4-VII and over-all system weights in Table I-4-VIII. Reduction of the super-orbital abort thrust to 12,000 lb would allow savings of approximately 300 lb in engine weight.

The four main thrust chambers are canted at 23 degrees to align thrust with the center of gravity, and each may be gimballed 5 degrees in any direction to follow center-of-gravity travel. This thrust is applied at four places on an octagonal ring which forms the main structural member of the propulsion system. All components except the attitude control system are mounted on the octagonal ring or on substructures attached to it. Thus, the principal components of the propulsion system are integrated in an assembly that may be acceptance tested, transported, and installed in the vehicle

TABLE I-4-VII. REVISED AEROJET-GENERAL PROPULSION SYSTEM NOMINAL DRY WEIGHT SUMMARY

	Unit Dry Weight, lb	Number Required	Total Dry Weight, lb
Fuel tank, outer, with insulation and mounts	335	1	340
Fuel tank, inner, with support cone	80	1	80
Fuel tank, auxilliary	22	1	22
Oxidizer tank, with insulation and mounts	54	4	216
Helium tank and supports	54	2	108
Thrust chamber assembly with propellant valves and gimbal actuators	128	4	512
Settling jets	10	2	20
Structure	121	1	121
Lines, fittings, valves, electrical	42	1	42
Attitude control thrust units	1	12	12
Attitude control tankage	25	1	25
Total Dry Weight, lb			1498

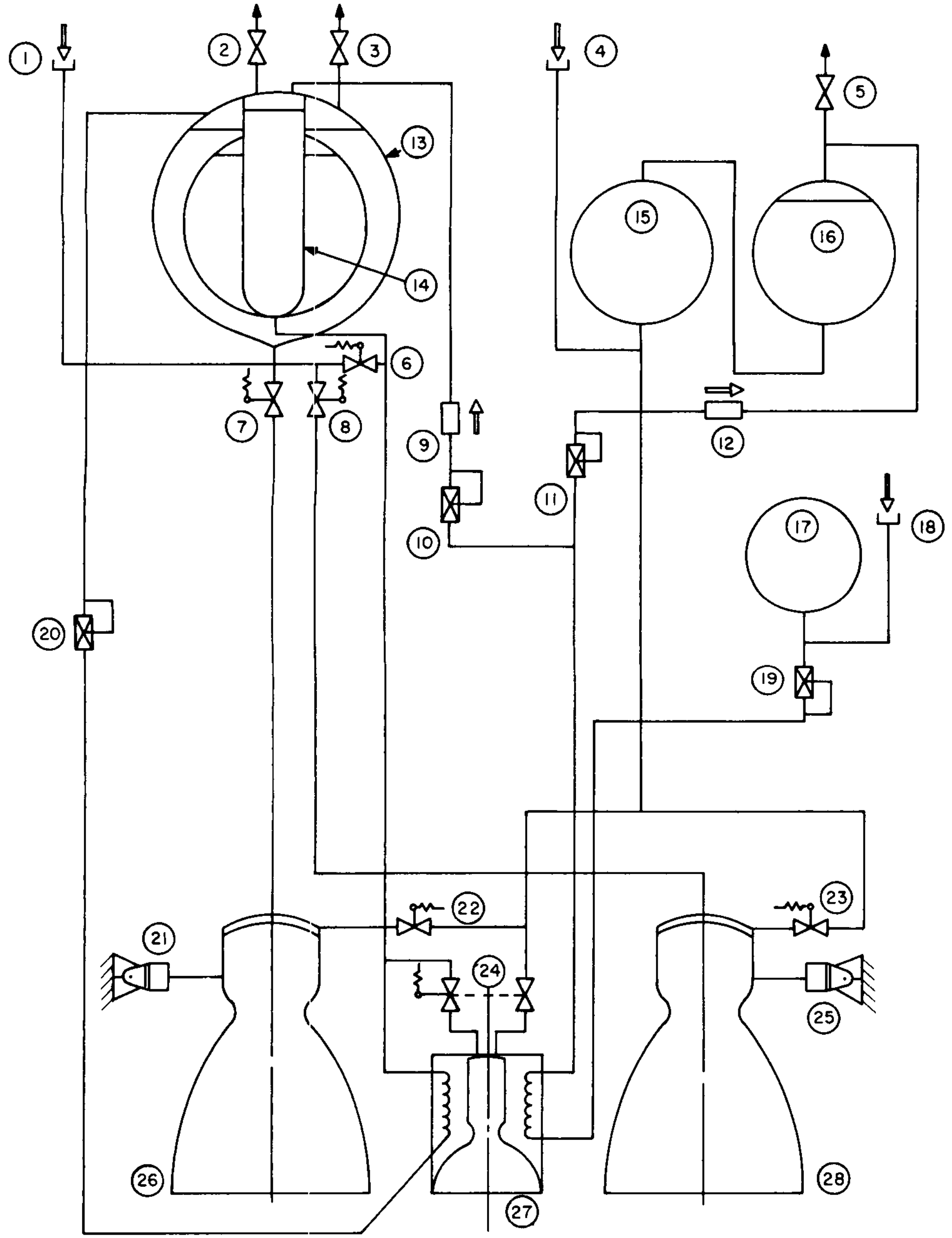
TABLE I-4-VIII. AEROJET-GENERAL APOLLO D-2 PROPULSION SYSTEM LOADED WEIGHT

	1963 System	
Powerplant Weight Summation		
Propellant		8,376 lbs.
Outbound midcourse	361	
Orbit maneuvers	7,562	
Inbound midcourse	219	
Attitude control	234	
Other Fluids		143
Fuel used for pressurization	120	
Helium	23	
System Dry Weight (including attitude control units)		1,498
Small Separation Rockets		43
Total Loaded Weight		10,060 lbs.

without disassembly or other operations which might disturb its proven operability. This same assembly may be left behind as a unit during the launch abort escape maneuver.

Envelope and heat transfer considerations dictate the use of a single spherical or near spherical hydrogen tank. To minimize length and remain within the specified envelope, the oxidizer was divided into four tanks spaced between the thrust chambers. This basic configuration is shown in detail in Figure I-4-8. A schematic of one-half of the propulsion system is shown in Figure I-4-10. The aft support structure is separated and left with the boost vehicle to leave the chambers free and to prevent impingement of the exhaust upon the aft skirt of the vehicle. Although not required for single thrust chamber operation, it may be necessary to provide a flame shield to restrict base recirculation and heating when all four thrust chambers are in operation. A suggested

[redacted]



Item	Description
1	Fuel Fill Disconnect
2	Auxiliary Fuel Tank Vent Valve
3	Fuel Tank Vent Valve
4	Oxidizer Fill Disconnect
5	Oxidizer Tank Vent Valve
6	Auxiliary Fuel Hydrogen Fill Valve
7	(1) Thrust Chamber Fuel Valve
8	(3) Thrust Chamber Fuel Valve
9	Auxiliary Fuel Pressurization Check Valve
10	Auxiliary Fuel Pressure Regulator
11	Oxidizer Pressure Regulator
12	Oxidizer Pressurization Check Valve
13	(1) Fuel Tank
14	Auxiliary Fuel Tank
15	Oxidizer Tank
16	Oxidizer Tank
17	(1) Helium Tank
18	Helium Fill Valve
19	Helium Pressure Regulator
20	Fuel Tank Pressure Regulator
21	(1) Thrust Vector Control Actuator's (2 Req'd)
22	(1) Thrust Chamber Oxidizer Valve
23	(3) Thrust Chamber Oxidizer Valve
24	Settling Jet Bipropellant Valve
25	(3) Thrust Vector Control Actuators (2 Req'd)
26	(1) Thrust Chamber
27	Settling Jet
28	(3) Thrust Chamber

Figure I-4-10. Lunar insertion propulsion system schematic

[redacted]

installation is shown in Figure I-4-11. Although ineffective as a radiation shield for the propellant tanks (which are already shielded), this device might offer some meteoroid protection and limit radiation to the supporting structure.

Using the method described in the Aerojet appendix, a parametric study was performed by Aerojet General to select the optimum levels of thrust chamber pressure, expansion area ratio, and propellant mixture ratio. The results of this analysis, based upon utilization of a Hylas Type pressurization system and four ablative thrust chambers, is presented in Figures I-4-12, I-4-13, and I-4-14. Figure I-4-12 (at a propellant mixture ratio of 5.0) shows the optimum thrust-chamber pressure to be a function of expansion area ratio with a nominal value of 70 psia at a 40:1 expansion. Optimum expansion area ratio, as shown by Figure I-4-13, is in the 40:1 to 50:1 range with little advantage for values over 40. Figure I-4-14 indicates optimum propellant mixture ratio to be just under 5.0. Selection of the Hylas design point of 65 psia chamber pressure, 40:1 expansion area ratio, and 5.0 mixture ratio as indicated on the curves (and at which considerable design and experimental work have been performed) represents almost exactly the optimum operating condition. Packaging considerations indicated that it was necessary to reduce the expansion area ratio to 35. Figure I-4-13 shows, however, that this does not result in an appreciable weight penalty.

4.2.2.4 DETAILED DESIGN FEATURES

4.2.2.4.1 Thrust Chamber Assembly

The Aerojet thrust chamber assembly consists of an ablative cooled combustion chamber and nozzle bolted to a lightweight aluminum injector. Thrust mounts and propellant valves are attached directly to the injector. The nozzle will be radiation-cooled between the area ratio of approximately 3:1 and the exit area ratio of 35. Table I-4-IX summarizes the AJ10-133 thrust chamber data and performance. Initial phases of development of the combustion chamber were completed during the Hylas program.

The Aerojet combustion chamber is constructed of an ablative liner, a thin layer of insulation, and a high-strength overwrap. This provides the high thermal resistance and the high strength needed for a lightweight design. The first 12 in of the ablative

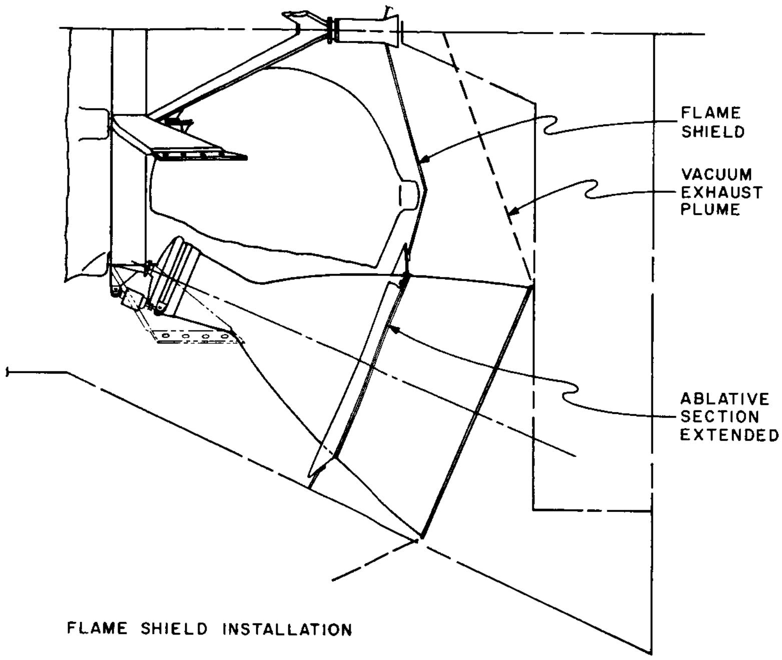


Figure I-4-11. Possible flame shield installation for AJ10-133 engine

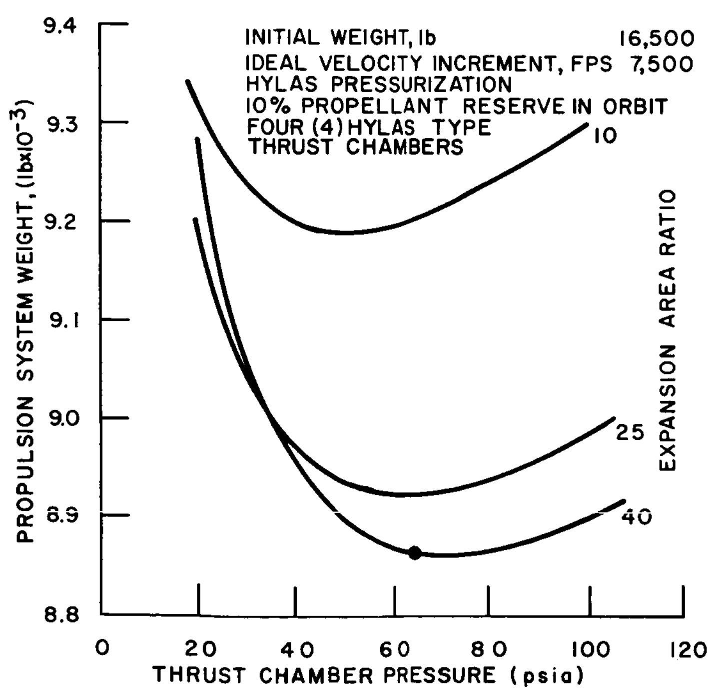


Figure I-4-12. Effect of thrust chamber pressure of propulsion system weight

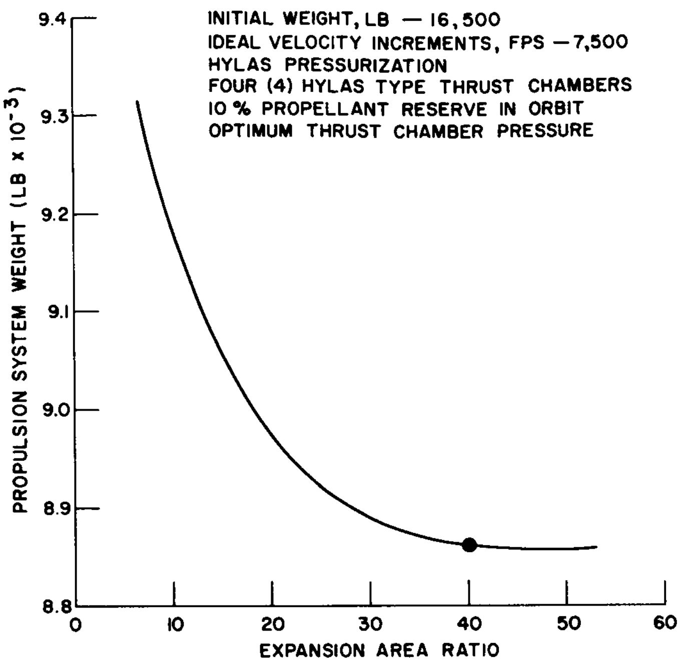


Figure I-4-13. Effect of expansion area ratio on propulsion system weight

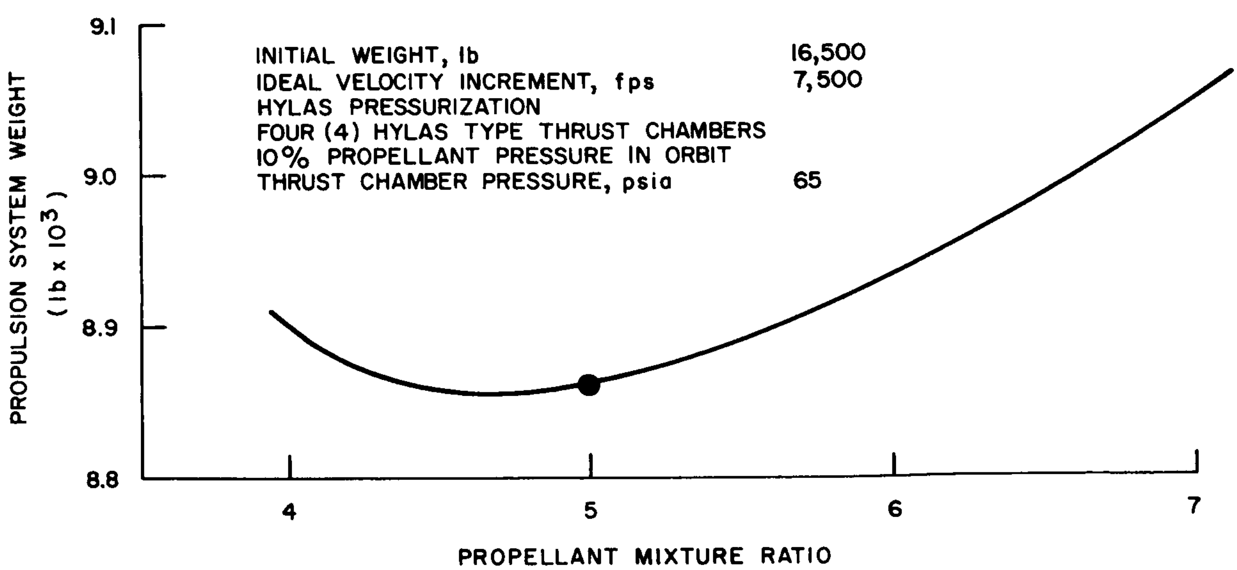


Figure I-4-14. Effect of propellant mixture ratio on propulsion system weight

TABLE I-4-IX. THRUST CHAMBER DATA

	Single Chamber	Four Chambers
OPERATING CONDITIONS		
Thrust (vacuum), lb	6,000	24,000
Propellants	LO_2/LH_2	LO_2/LH_2
Chamber Pressure, psia	65	65
Propellant Flow Rate, lbm/sec	13.95	55.8
Mixture Ratio	5:1	5:1
Expansion Area Ratio	35:1	35:1
Specific Impulse (vacuum), sec	430	430
Maximum Total Duration of Full Thrust, sec	546	137
DIMENSIONAL DATA		
Overall Length, in.	61.3	-
Exit (outside) Diameter, in.	50.0	-
Throat (inside) Diameter, in.	8.05	-
Contraction Ratio	2:1	-
MATERIALS		
Injector	Aluminum	-
Combustion Chamber	Ablative Plastic Fiberglass wrapped	-
Expansion Nozzle	Titanium	-

liner is composed of phenolic-impregnated asbestos fibers, edge-wrapped with a 60-degree orientation to gas flow. The area from 12 in below the injector to 5 in below the throat is 60-degree edge-wrapped Refrasil (phenolic-impregnated quartz fibers), and the nozzle portion from 5 in below the throat to an area ratio of 3:1 is the same type asbestos wrap as the upper chamber. A thin wrap of tangentially oriented phenolic-impregnated asbestos is used on the outside of the Refrasil portion for insulation. The high-strength overwrap of the entire assembly is composed of glass cloth for longitudinal strength and circumferential-wound glass filaments for hoop stress. The glass wrap is bonded with epoxy resin. The high thermal resistance of the ablative liner, plus the asbestos insulation behind the Refrasil, isolates the outer wrap and permits it to be used at moderate temperatures where strength is high. The use of nonmetallic materials at moderate temperatures (300 F) in vacuum conditions for periods of 30 days has been shown to be no problem.* Specimens subject to these conditions have shown a 1-2 percent decrease in ablative material weight and a very slight loss in flexural strength. Similar control specimens subject to the same temperature history but at sea level pressures show similar changes in properties substantiating the theory that with chain polymers the temperature rather than the vacuum is the rate controlling factor and the process is one of pyrolysis rather than evaporation or sublimation.

Following shutdown of an ablative thrust chamber after a long-duration run, the chamber will continue to ablate until it cools below the ablation temperature. The method of Appendix P-A shows that this required approximately 30 seconds and, for the Aerojet chamber, will result in a char depth growth of approximately 10 percent. Thus, any reasonable number of restarts can be designed for by selecting a suitable thickness of ablative material. The chamber recommended for this application is capable of up to 17 firings. On short duration runs such as may be required for course corrections, the heat sink capability of the chamber may not be exceeded and the ablation process not started. See Appendix P-A.

*Research and Development on Components for Pressure-Fed Liquid Oxygen-Liquid Hydrogen Upper Stage Propulsion Systems, Report No. 1933 (Final) Contract AF 0416 (616)-5170, Aerojet General Corporation, Azusa, Calif.

The ablative material will be terminated at an area ratio of 3:1, and a radiation-cooled metallic skirt will be attached through a bolt-on flange. The mass of the flange is sufficient to avoid an excessive temperature rise with the resultant bonding problems. A trapped O-ring seal is used to provide for convenient assembly of the thrust chamber and nozzle at the launching or test site.

When more than one thrust chamber of the cluster is in operation, cross radiation between nozzle expansion skirts will take place raising the skirt temperature. The most critical condition exists on the portion of the nozzle nearest the vehicle centerline when all four thrust chambers are in operation. Due to the relatively wide spacing of the thrust chambers and the fact that the exhaust plume is transparent to radiation from the nozzle skirt, the solid angle viewed by a nozzle element at this location is reduced by only 19 percent. The resulting 5-percent rise in temperature is readily compensated for in the design.

Test firings at Aerojet of radiation-cooled nozzle extensions with clusters of 1/16-in holes drilled at area ratios of approximately 10, 15, and 25 have been conducted to verify that skirt integrity will be maintained in the event of a meteoroid puncture. Post fire examination of the skirts after tests of 30 seconds duration at a chamber pressure of 150 psia revealed no apparent growth of the holes.

Two injector configurations are envisioned by Aerojet for the experimental phase; one is a conventional, concentric-ring, shower-head design, and the other is a design containing a multiplicity of rosettes in a face lined with ablative material. In both designs, intermanifold welds are minimized, and rapid breakup of the oxidizer is emphasized. This latter operation has been shown experimentally to be the key factor in achieving high performance with LO_2/LH_2 propellants. A simple "mono-ball" structure is used for thrust take out. This design permits easy accessibility for servicing.

Ignition is accomplished in the Aerojet chambers by four surface-gap spark plugs located around the periphery of the injector. These plugs are positioned such that the injector film cooling will protect them during steady-state operation. During the starting sequence, a 0.1 sec oxidizer lead is programmed to provide oxidizer in the area of the plugs at the time fuel flow starts. Tests have proven this lead time to be adequate for ignition to occur before the fuel film blankets the plugs. This system has been developed by Aerojet and proved in over 30 firings on Titan-size hardware using LO_2/LH_2.

Estimated start and shutdown transients of the AJ10-133 engine are given respectively in Figures I-4-15 and I-4-16. The start transients, as shown in this curve, are based on pressurized tanks. For initial runs of the system when the ullage is small, pre-pressurization of the tanks can be accomplished in 1 to 2 seconds. This would be the situation in the event of a super orbital abort. Later runs, where the tank ullage is high, might require several seconds pressurization time.

Figure I-4-17 shows the degradation in performance associated with short-duration runs due to the inefficiency of the start and shutdown transients. These data are based upon an average of several Aerojet Hydra-Hylas test runs which indicate an effective specific impulse of 340 sec (corrected to vacuum) during the start and shutdown periods.

4.2.2.4.2 Pressurization System

For propellant pressurization, the AJ10-133 system utilizes hydrogen to pressurize the fuel and helium to pressurize the oxidizer. This system has four principal components: An auxiliary fuel tank, a helium-sphere, a heat exchanger, and a settling rocket. The design parameters used have all been verified by the Hylas test program in over 40 expulsion tests.

To provide a positive pressure differential between the supply of pressurization fluid and the fuel tank, a pressurized auxiliary tank is used. Hydrogen is stored as a liquid in the auxiliary fuel tank to keep the volume and weight of the tank to a minimum. This is accomplished by submerging the auxiliary tank in the main fuel tank, which also saves space and eliminates the need for insulation. The liquid hydrogen is supplied to the heat exchanger by helium pressurization of the auxiliary fuel tank. The use of helium for this application does not present any problems, because the density of the helium at the design temperature and pressure (38 R, 185 psia) is less than the density of liquid hydrogen under the same conditions.

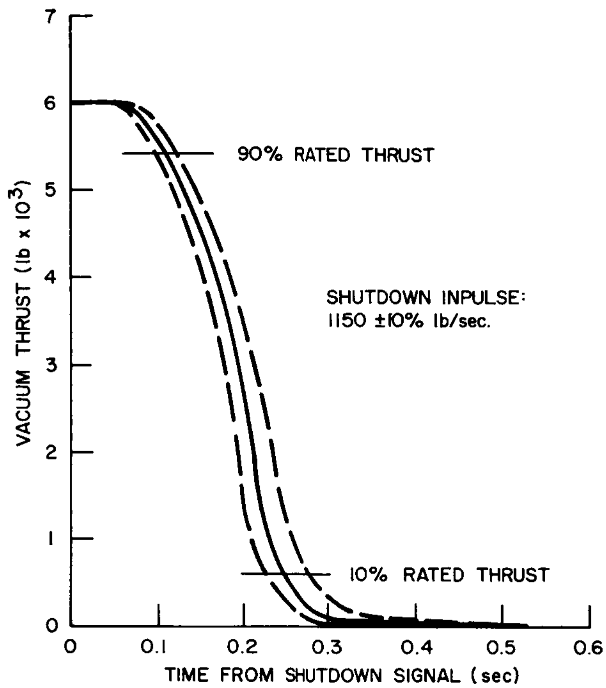


Figure I-4-15. Estimated start transient

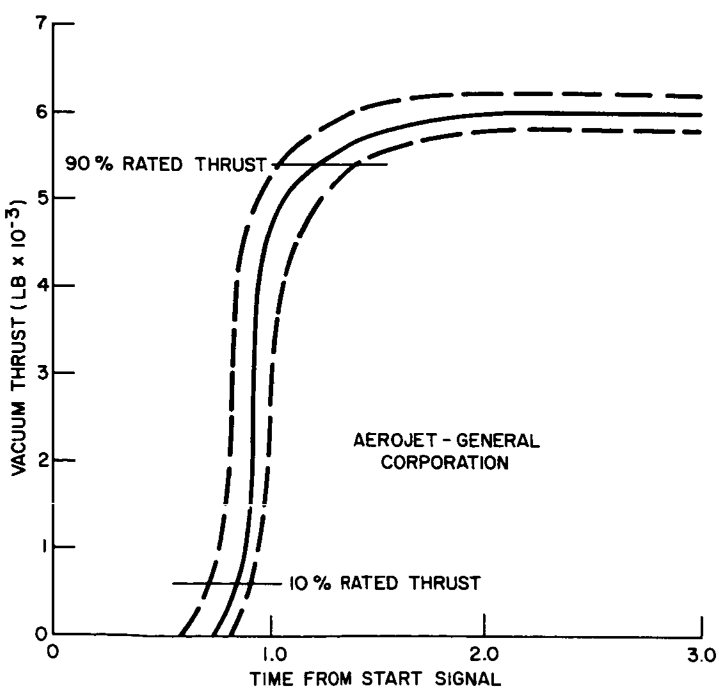


Figure I-4-16. Estimated shutdown transient

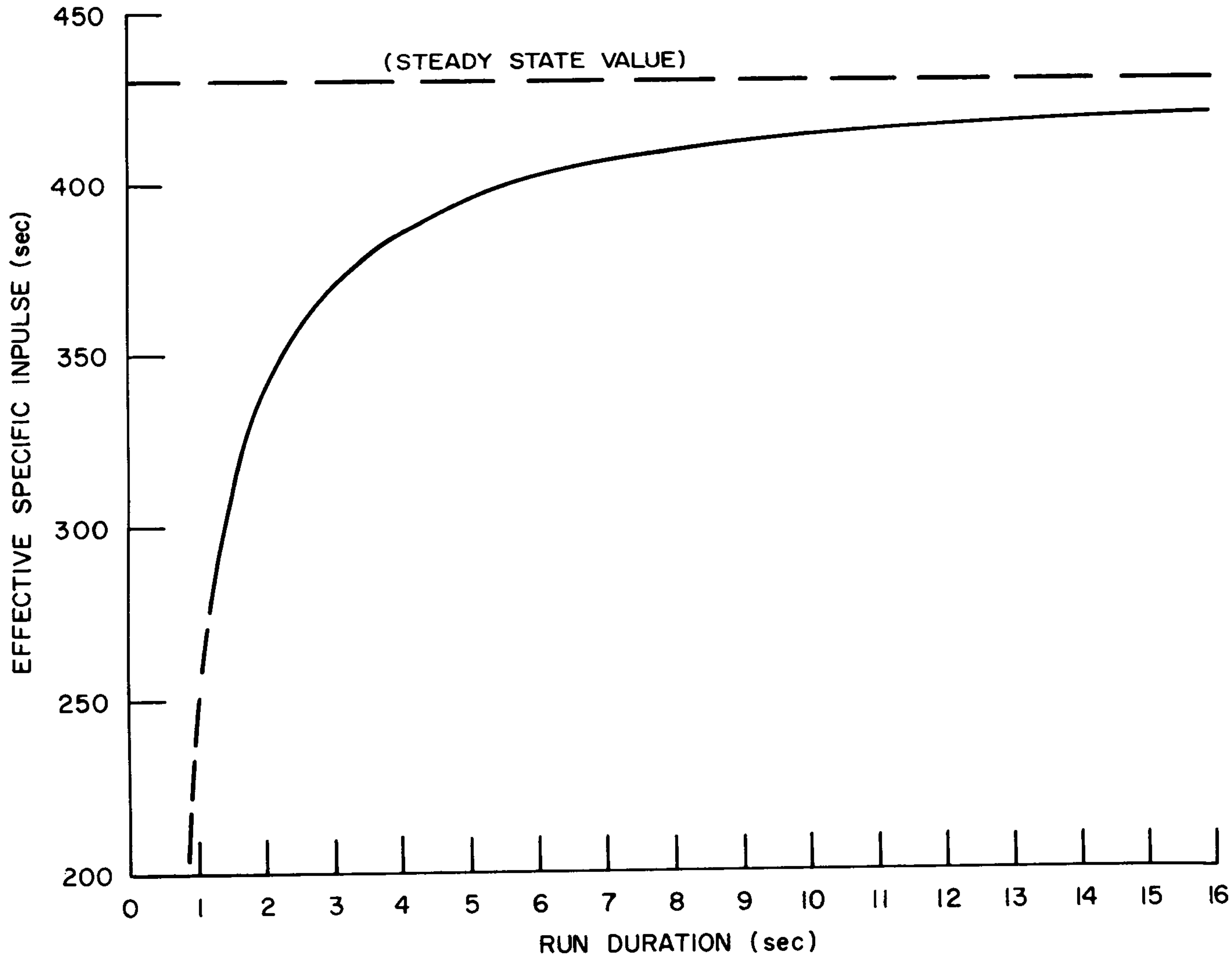


Figure I-4-17. Effective Specific impulse for short duration (course correction) firings

4.2.2.4.3 Tankage and Structure

A titanium alloy was chosen for the liquid hydrogen tanks which consist of an outer spherical tank, an inner spherical tank, and a cylindrical auxiliary tank supplying hydrogen for pressurization. The alloy (A110-AT) may be readily formed and welded, has a high strength/density ratio without heat treat, and has good impact strength at -423 F.

Titanium is not proposed for use in the liquid oxygen tanks because of questionable compatibility. Previous experience has indicated such usage might be hazardous. Instead, a heat treatable aluminum alloy 6061, is used for the liquid oxygen tanks. Two hemispheres are fabricated and heat treated to the T6 condition. The helium storage sphere is installed, and the hemispheres are inert arc gas welded together. Wall thickness at the girth weld is great enough to reduce stress below yield in the heat-affected area. A heat-treatable alloy was used instead of depending on work-hardening for high yield strength because of the several bosses and attachments which may conveniently be welded-on before heat treat. A material with a higher strength/density ratio, such as heat treated AM350 or 17-7 PH, was not used because the tank wall thickness is already at the minimum for handling loads with the aluminum.

The two helium storage spheres for the main propellants are fabricated from AM350, heat-treated to a room temperature yield strength of 135, 000 psi. The two helium tanks immersed in liquid oxygen have a yield strength of 190, 000 psi. This material is compatible with the oxygen, may be welded and machined before heat treat, has a high strength/density ratio especially at cryogenic temperatures, and has sufficient ductibility at the temperature of liquid oxygen. A summary of tank data is included in Table I-4-X.

The octagonal ring which constitutes the principal member of the propulsion system frame is supported by eight attachments to the vehicle structure. The ring, in turn, supports the hydrogen tanks, the four oxygen tanks, the four main thrust-chambers and a sub-frame on which the settling jets are mounted. The frame utilizes box-beam construction and is fabricated from 7075-T6 aluminum alloy sheet and extrusions. Its weight is 121 lb including all attachments.

TABLE I-4-X. TABULATED TANK DATA

Tank	Diameter, in.	Shape	Operating Pressure psia	Material	Wall Thickness in.	Number Used and Total Dry Weight, lb.
Hydrogen #1	100	Spherical	100	Ti A110-AT	0.025	1, 151
Hydrogen #2	75	Spherical	100	Ti A110-AT	0.020	1, 66
Hydrogen Aux.	18	Cylindrical (92 in. long)	200	Ti A110-AT	0.020	1, 22
Oxygen	44	Spherical	100	A1 6061-T6	0.033	4, 142
N_2O_4/A-50	17.5	Spherical	200	A1 6061-T6	0.040	2, 10
Helium	18	Spherical	4,000	Steel Am 350	0.150	2, 104
				Total Tank weight, lb (dry)		495

The use of eight points of attachment to the vehicle structure, a relatively high number, is compatible with the number of components which it supports and is structurally sound. The eight attachments provide good load distribution in the vehicle and low bending moments in the octagonal ring which is essentially an assembly of eight simple beams. At each corner of the octagon, a short radial beam spans the distance to the corresponding attachment point to the vehicle. The attachment points are located on a 117-in diameter circle. Each attachment transmits a maximum shear load of 6,000 lb and a maximum moment of 87,000 in-lb to the vehicle.

The applicable tank data are tabulated in Table I-4-X.

4.2.2.4.4 Thrust Vector Control Actuators

The AJ-10-133 engine uses thrust vector control actuators to allow thrust vector alignment through the vehicle center of gravity. Previous studies indicate that an electric motor servo mechanism with a ball-screw actuator is suitable for operation at very low temperatures such as are encountered in an O_2/H_2 system. Work is in progress at Aerojet on actuators for similar applications. Therefore, their use is considered feasible here.

4.2.2.5 COMPONENT STATUS SUMMARY

A brief description and status summary of major components are presented in Table I-4-XI.

4.2.2.6 MALFUNCTION DETECTION AND SEQUENCER UNIT

4.2.2.6.1 Purpose

The malfunction detection and sequencer distributes electrical power to control operation of the motors and engines. It can be designed using state-of-the-art principles similar to those used in the Malfunction Detection System for the Dyna-Soar engines presently being designed and the XLR91 (Titan State II) Airborne Sequencer.

Engine parameters can be monitored to detect incipient engine failure. These parameters will be used as criteria for engine shutdown and also initiate redundant equipment start up.

TABLE I-4-XI. SUMMARY OF COMPONENT STATUS AT AEROJET-GENERAL

Component	Description	Remarks
Combustion Chamber	Ablative cooled low pressure	Feasibility and design criteria established by Aerojet under AF 33(616)-7401 and AF 04(611)-5170 (Hydra-Hylas)
Expansion Nozzle	Radiation cooled titanium	Demonstrated on simulated altitude tests and five flights of Ablestar vehicle. Puncture tests showed skirt insensitive to meteoroid damage.
Injector	Concentric ring showerhead	High performance demonstrated by Aerojet on Hydra-Hylas test program.
Propellant Valves	Butterfly type	Scaled down versions of 1-203990 fuel valve and 1-203650 oxidizer valve used on Titan and approximately 55 oxygen hydrogen firings
Ignition System	Surface gap spark plugs	Use of 1 to 4 spark plugs demonstrated by Aerojet on Hydra-Hylas test program. Research indicates 0.05% O_3F_2 (wt % in O_2) may promote reliable hypergolic ignition.
Thrust Vector Actuators	Electric motor and ball jack-screw	Since high gimbal rates are not required, a simple system is possible utilizing commercially available parts.
Settling Jet-Heat Exchanger	Low pressure, low mixture	Performance not critical. Design criteria established on Titan and Hydra-Hylas programs. 1-234290 fuel and 1-234476 oxidizer (Titan gas generator valves) valves can be used.
Tankage	Spherical, conventional	Good background at Aerojet in flight weight pressurized propellant tanks. Experience and facilities from Able, Delta, Ablestar, Aerobee, and others applicable.
Structure	Box beam or tube, conventional	Standard structural frame techniques used.
Gas Pressure Regulators	Commercial products	Hadley P/N 10998 (hydrogen) and Skyvalve (helium) P/N R0102-8P performed satisfactorily in 40 pressurization system tests.

Logic and timing for all phases of the flight except re-entry spin control can be contained in this unit. Temperature control of this device which may contain semiconductors could be obtained by installation in the mission module or by installation on the vehicle wall.

4.2.2.6.2 Electrical Power Requirements

Power input can originate from a single or dual (redundant) source. For operation of one thrust chamber, approximately 2 amp at 28 vdc and 5 amp at 115 v, 400 cps are required. For super orbital abort, approximately 4 amp at 28 vdc and 13 amp at 115 v, 400 cps are required. The duration of these requirements is only about half that for the normal mission. Attitude control thrust chamber valves require 0.5 amp each at 28 vdc. Since there are twelve such valves (any six of which could be operated at one time), up to 3 amp at 28 vdc could be used.

If minimum energy consumption is desired, the thrust chamber igniters may be turned off after ignition.

4.2.3 Propulsion System Operation

The selected system would use the following sequence of events: *

4.2.3.1 NORMAL

(1) Escape and high dynamic pressure separation rockets are jettisoned; four at first-stage burnout, six at second-stage burnout, and two at third-stage burnout.

(2) After boost, but before midcourse correction, attitude is automatically corrected by the attitude control system (ACS) which functions as needed for the duration of the flight up to re-entry unless over-ridden by the pilot.

(3) Fuel gages and all tank pressures are checked to ensure that the propellant system is normal.

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* Where engine restart is involved, the steps for restart are omitted for simplicity.

(4) All engines are checked for continuity at this time. (This may not be necessary, but is suggested for consideration.)

(5) All other checks of vehicle normalcy (electrical power supply for engine, etc.) are made at this time.

(6) #(1) Thrust vector control actuator is energized to move the engine to nominal firing attitude to minimize "kick" at start-up.

(7) Propellants are settled with the (1) settling jets. Thrust chamber igniters are turned on.

(8) #(1) Thrust chamber fuel and oxidizer valves are opened.

(9) The propellants ignite and burn, and thrust is supplied until the guidance computer determines that the velocity vector is correct.

(10) #(1) Thrust chamber fuel and oxidizer valves are closed.

(11) #(1) Thrust chamber igniters are turned off.

(12) During coast to the vicinity of the Moon, Steps 3, 4, and 5 are repeated as necessary.

(13) The engine is again fired using the procedure in Steps 6, 7, 8, 9, 10, and 11 to accomplish lunar insertion.

(14) The No. 1, lower, outer, LH_2 tank is vented to space to prevent pressure buildup and inner tank collapse.

(15) During the stay in orbit around the Moon, Steps 3, 4, and 5 are again repeated.

(16) The No. 2 engine is fired using a procedure similar to Steps 6, 7, 8, 9, and 10, except that No. 2 hardware is used to accomplish the lunar exit maneuver.

(17) During the coast back to the vicinity of the earth, Steps 2, 3, 4, and 5 are repeated for the final midcourse maneuver.

(18) The No. 2 engine is again fired to accomplish the midcourse correction. The main propulsion module is disconnected.

(19) At the correct point in relation to the earth, the separation rockets are fired to separate the re-entry vehicle from the spent spacecraft.

(20) Any spin of the re-entry vehicle is automatically corrected by the re-entry spin-control jets.

4.2.3.2 ABORT DURING BOOST OR ON THE PAD

(1) The booster malfunction detection system detects a booster malfunction necessitating abort.

(2) The propulsion system attachment bolts are fired.

(3) Aerodynamic drag on the lower section of the skirt separates the vehicle and main propulsion.

(4) The eight solid rocket abort motors are fired.

(5) The APOLLO vehicle (less main propulsion) is accelerated away from the Saturn booster for two seconds.

(6) The spacecraft aft shell is separated.

(7) The high dynamic pressure separation rockets are fired to separate the re-entry vehicle from the spacecraft.

(8) Any spin of the re-entry vehicle is automatically corrected by the re-entry spin control jets.

4.2.3.3 SUPER-ORBITAL ABORT

(1) After separation from the third stage, super-orbital abort is possible if immediate return to earth is required. This can occur any time after orbital velocity is achieved during third-stage burning. If the booster is not separated, the first step is to fire the two remaining abort rockets to escape from the third stage. The main propulsion module is retained in this situation.

(2) A decision is supplied on the most appropriate super-orbital abort maneuver.

(3) Initiate super-orbital abort.

(4) The attitude control orients the vehicle to the proper attitude.

(5) Steps 6, 7, 8, 9, and 10 of paragraph 4.2.3.1 are automatically performed except that engines 1, 2, 3, and 4 fire simultaneously to produce a 24,000-lb thrust.

(6) The vehicle is deflected and heads toward the atmosphere.

(7) Steps 18 and 19 are performed to accomplish re-entry.

4.2.3.4 ABORT AFTER BOOST BUT BEFORE LUNAR INSERTION

(1) During steps 1 through 7 of the normal sequence of events (paragraph 4.3.2.1), an uncorrectable situation is discovered. Procedure is normal except redundant equipment is used.

(2) The pilot decides to abort the attempted circumlunar mission and make a free return to earth (cislunar mission).

(3) If there is no danger of impacting the Moon, midcourse correction is delayed until after apogee is attained.

(4) Normal sequence is resumed starting with Step 16.

4.2.3.5 MALFUNCTION AFTER LUNAR INSERTION

(1) Procedure is Normal using redundant equipment.

4.2.4 Space Storage of Propellants

Perhaps the key to successful utilization of cryogenic, high-energy propellants is the successful storage and expulsion during the 14-day mission. Heat leaking into the propellants must be minimized by minimizing tank surface area and using good insulation (such as Linde SI-4 plus utilization of the vacuum of space), suitable tank supports, and an adequate pressurization system. Propellant tank venting to relieve the pressure built up by this heat is difficult to achieve for this mission and wastes propellant energy. Proper design and insulation of tanks should minimize the total pressure which can be kept well below 100 psi. Since minimum gage problems dictate that walls will stand at least 100 psi, it is not planned to vent the cryogenic propellant tanks during the mission.

Available space permits storage of hydrogen in a single exposed sphere. This provides minimum weight for the largest volume tank and minimum surface area to insulate. The liquid oxygen will be stored in four individual spheres.

Highly efficient lightweight insulations are commercially available which are suitable for space storage. Linde Type SI-4 has been tentatively selected as being representative of the multiple-radiation-shield type of insulation. It consists of 40 to 80 layers of aluminum foil per inch separated by submicron glass fiber paper. When the pressure of the insulating space is at 1 micron of mercury or less, the insulation has very low thermal conductivity. It has no structural strength, but will support its own weight under considerable vibration and shock loading. The aerodynamic shield used to stabilize the spacecraft during the early abort phase will serve to protect the insulation during boost. Sufficient studies have been conducted to conclude that this insulation will be adequate for the mission without a severe weight penalty. However, a detailed study is needed to determine the optimum insulation thickness. A thickness of 2 in on the hydrogen tank and 1/2 in on the oxygen tanks was selected for the preliminary design. The heat transfer rates and weights of insulation are shown on Figures I-4-18 and I-4-19. For these curves, it was conservatively assumed that the outer layer of insulation was at 530 F. The resulting heat transfer rates are probably somewhat high, since the outer layer of insulation will face other cold propellant tanks and structures as well as the warm outer skin of the vehicle.

The structural design of a vehicle using O_2/H_2 propellants has a great effect in determining the adequacy of the vehicle for space storage. Even if a highly effective insulation is used to reduce the amount of external energy absorbed, heat conduction through structural members can negate the effect of this insulation. Also, the structural members can serve as easy paths for heat from the various internal sources such as the payload and guidance and control units.

A common method of reducing the heat transfer to cryogenic fluids is to suspend the tanks on long, highly stressed tensile members. Because of the specific design requirements of the APOLLO, it is not possible to use tensile members without imposing a severe weight penalty. Therefore, a "heat barrier" system is used which employs the principle of a series of stacked plates, forming a laminated, multiple-contact

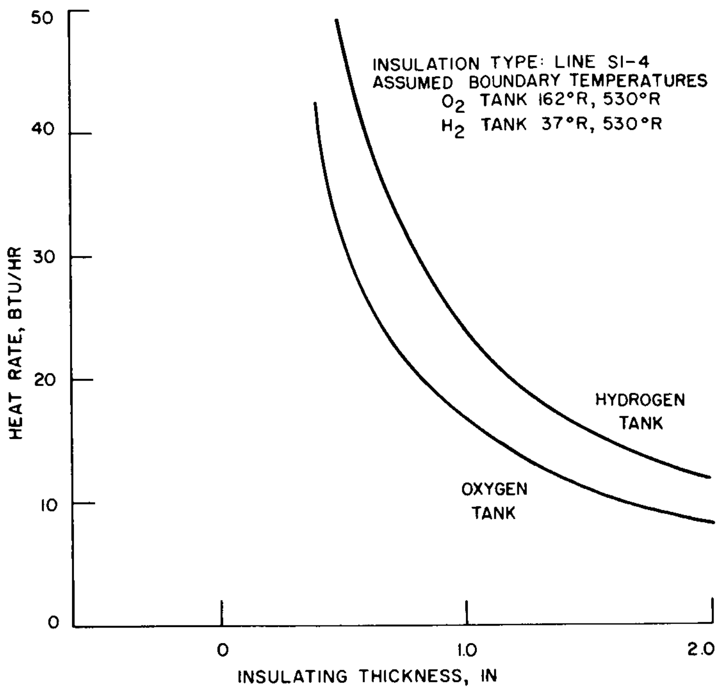


Figure I-4-18. Heat transfer rate vs insulation thickness

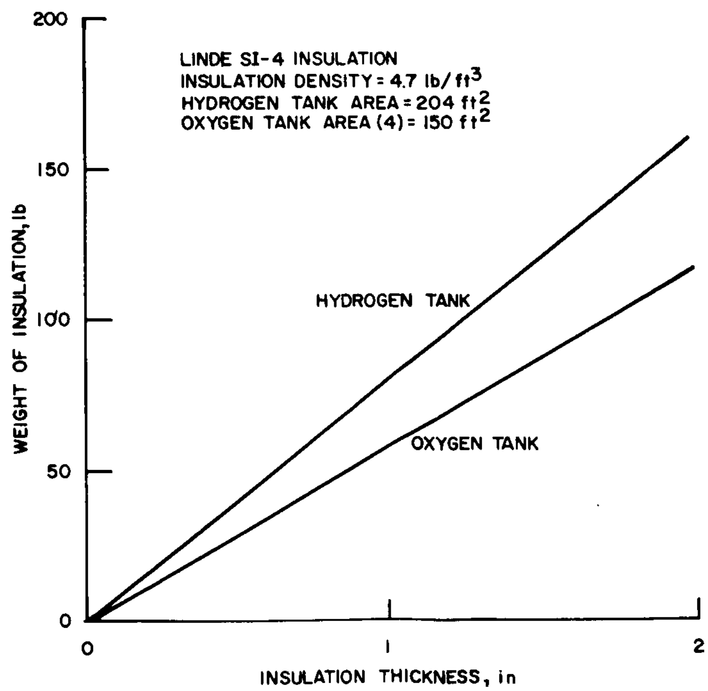


Figure I-4-19. Insulation weight vs thickness compression support member. The effectiveness of this principle has been demonstrated.* The thermal resistance of the gap between two pieces of metal pressed together increases the thermal resistance of the member without reducing its compressive strength. The resistance of the gap may be further increased by dusting the plates with manganese dioxide, or by placing layers of Micarta between the metal plates. The actual configuration selected for the preliminary design utilizes a metal strip tightly rolled into a coil. A typical curve of the heat current through a member of this type is shown in Figure I-4-20. Since there is no load on the coil during the coast periods, the heat transfer will be low during these periods.

After the heat transferred to the tanks has been minimized, three methods of storage are possible: Storage in an unvented tank with a refrigerator to reliquefy the propellant boiloff, storage in a vented tank, and storage in unvented tanks, allowing the temperature and pressure of the propellants to rise. Storage by refrigeration was considered by Aerojet briefly and found to be undesirable for the low heat rates and short storage times of the APOLLO vehicle. Therefore, this method was not considered further.

The simplest way of storing cryogenic propellants is to utilize the heat capacity of the propellants by allowing the temperature and hence the vapor pressure to rise. By utilizing this method, the problem of venting the propellants in a gravity-free condition is circumvented, and no additional propellants must be carried along to compensate for losses due to venting. However, a decrease in density and stratification of the propellants may occur with diffusion and/or conduction of energy into the propellants being the main mechanism of heat transfer. At high rates of heat transfer, a vapor envelope may tend to form resulting in a reduced heat capacity of the storage system for a given pressure limit of the tank, because the bulk temperature of the fluid will not rise uniformly with that of the gas. The vapor pressure of the fluid would then be below the tank pressure. However, the vapor envelope itself would form a heat barrier which would reduce the rate of heat transfer to the tanks. Even without stratification or formation of a vapor envelope, the propellants for the lunar mission return trip midcourse corrections will undergo a considerable vapor pressure rise. This may be attributed to the small mass and hence low heat capacity of the propellants required. The tank

* Heat Conduction Through Insulating Supports in Very Low Temperature Equipment, R.P. Mikesell and R.B. Scott, Journal of Research, NBS Research Paper #2726, Vol. 57, No. 6, dtd Dec 1956

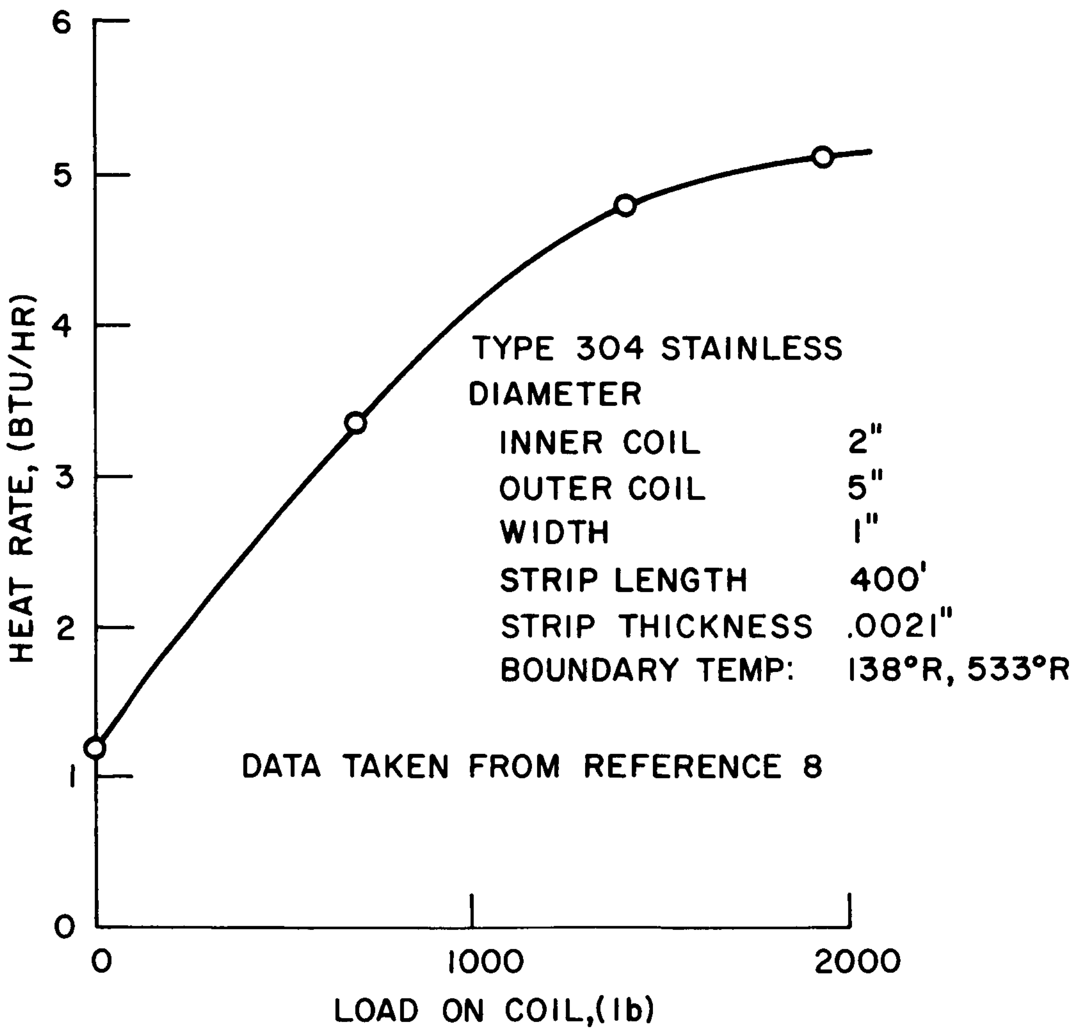


Figure I-4-20. Heat rate vs load on laminated support pressure schedules and temperature for a Hylas-type pressurization system have been calculated for a storage heat of 20 Btu/hr into the hydrogen tank and 50 Btu/hr into the oxygen tanks. The results are shown in Figures I-4-21 and I-4-22.

Operation of the Hylas-type pressurization system is described in Appendix P-A. Since propellant density is a function of temperature, the densities of both propellants will decrease during the storage period. This will cause a decrease in propellant flow rates and a shift in thrust chamber mixture ratio. The calculated mixture ratio is shown in Figure I-4-23. The rise in temperature and pressure of the propellants after a firing is due to the heat added by the pressurizing gas. It was assumed that after each firing sufficient time existed for the pressurizing gas and the remaining liquid to come to thermal equilibrium. If this does not occur, less heat will be absorbed by the liquid, and less shift in mixture ratio will result. Some form of flow-regulating device could be used to maintain the mixture ratio at a preselected value. However, its use degrades system reliability, and it is felt that a more realistic approach is to let the mixture ratio vary and accept the small degradation in performance.

An alternate pressurization system for return from space uses the vapor pressure for self expulsion of the propellants. This provides a type of redundancy in this critical area of pressurization.

The tank pressure history for a VaPak type pressurization system is shown in Figure I-4-24. The operation of the system is described in Appendix P-A. In this system, the energy to expel the propellants is obtained from the heat stored in the propellants, by allowing the propellant temperature (and hence vapor pressure) to drop during the run. The pressure drops during firing have been computed and compared with values* determined by Linde and shown in Figure I-4-25. Testing is currently being conducted to substantiate the computed values. The tank pressure and propellant density variations during a firing result in a larger shift in mixture ratio than in a Hylas-type system where tank pressure throughout a firing remains constant. The mixture ratio variation for the lunar mission is shown in Figure I-4-26, and the resulting specific

* Pressure Phenomena During Transfer of Saturated Cyrogenic Fluids, J.M. Canty, presented at 1960 Cyrogenic Engineering Conference, Linde Company, Division of Union Carbide Corp., New York, N.Y.

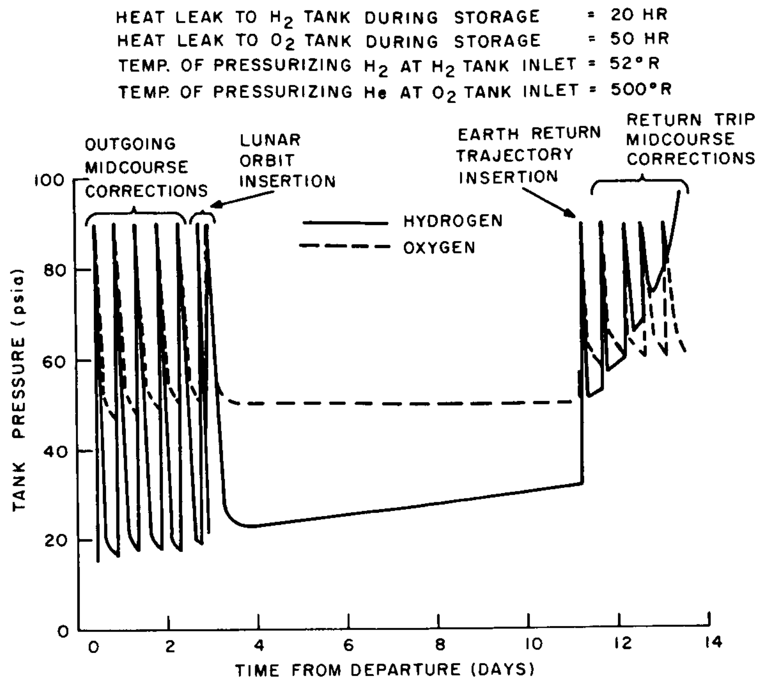


Figure I-4-21. Tank pressure vs time from departure - hylas system

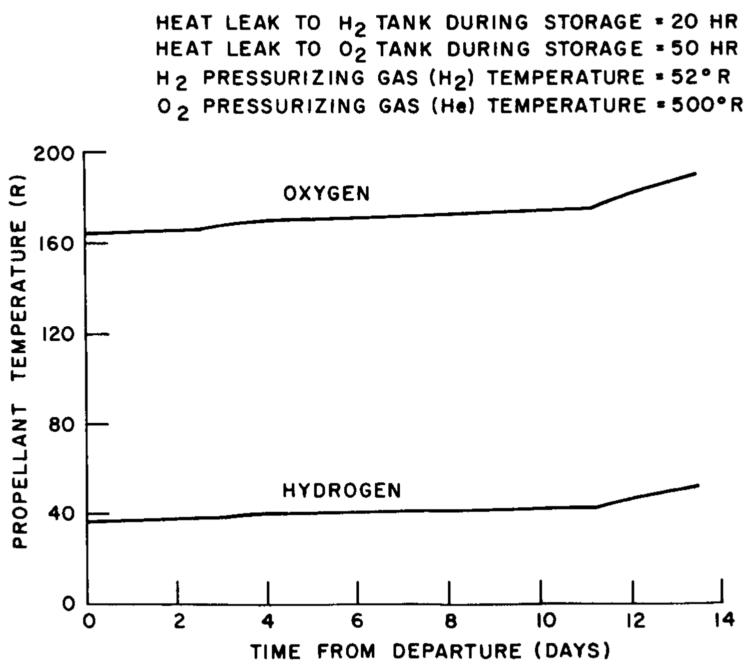


Figure I-4-22. Propellant temperature vs time from departure - hylas system

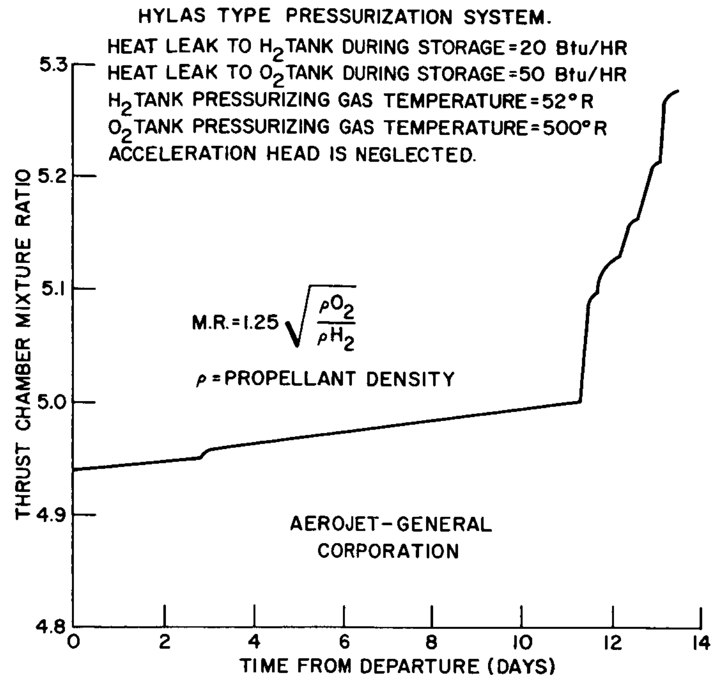


Figure I-4-23. Propellant mixture ratio vs time from departure - hylas system

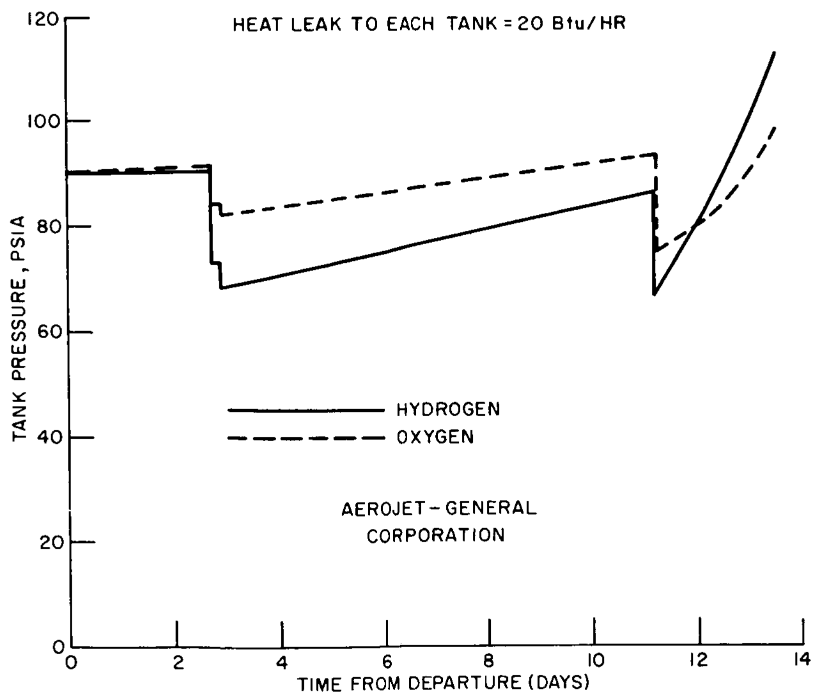


Figure I-4-24. Tank pressure vs time from departure - VaPAK system

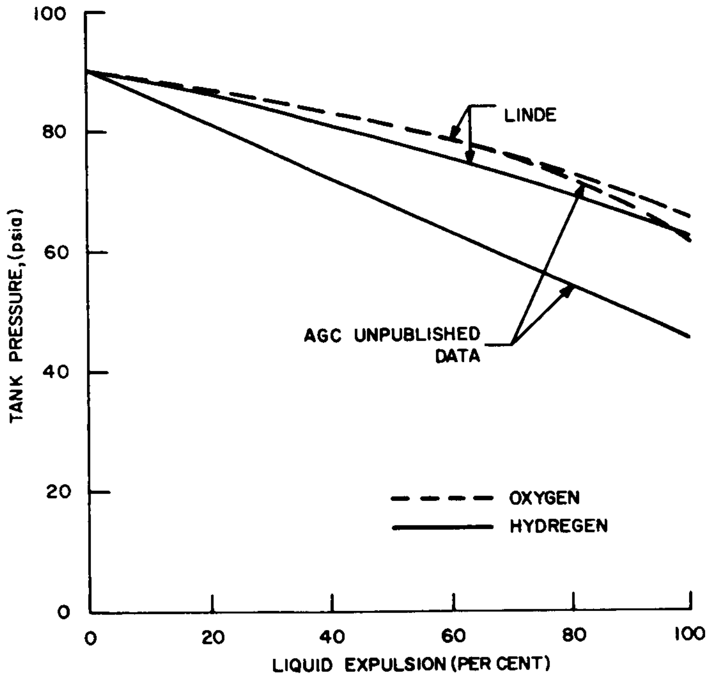


Figure I-4-25. Pressure decay comparison for VaPAK system

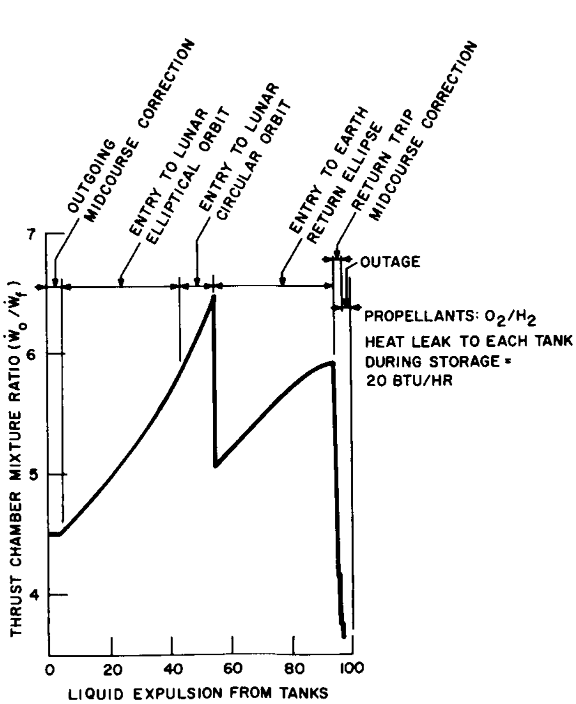


Figure I-4-26. Mixture ratio for APOLLO lunar mission with VaPAK pressurization

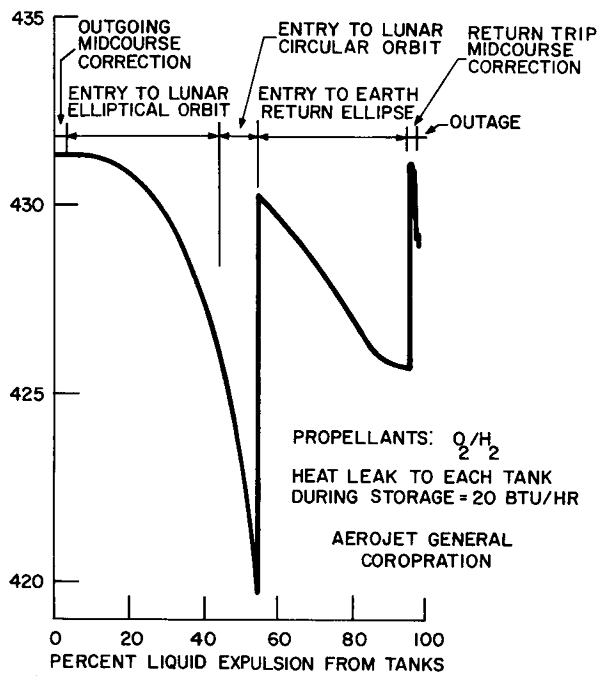


Figure I-4-27. Specific impulse for APOLLO lunar mission with VāPAK pressurization system impulse variation is shown in Figure I-4-27. It may be necessary to use a flow-regulating device in the VaPak system. However, the system is inherently reliable because of its simplicity, and the heat leak to the tanks will be low because no pressurization plumbing or auxiliary equipment is required.

4.2.5 Reliability and Safety Apportionment for AJ-10-133 Engine

Reliability is defined as the probability that the propulsion system will operate successfully, so that orbit about the Moon and return to earth is possible.

The system considered here is an integrated liquid rocket system using solid rockets for abort and separation maneuvers.

The only way the mission can be accomplished is to have no failure during the boost phase. After boost, one engine failure of each lunar maneuvering pair and one tank failure can be survived. Failure of the remaining tank cannot be survived.

After lunar orbit insertion, one engine failure can be survived, but no failure of the remaining tanks is permissible. If the malfunction detection system fails when it is needed, the mission fails unless the pilot and observers on earth can be used as a redundant malfunction detection system. The malfunction detection system could be of either the fail-run or fail-safe type. If the fail-run type is used, the malfunction detection system would shut the No. 1 engine down if it detected a "self" failure.

Assumptions made in the analysis that follows are that failure of a tank or engine does not induce failure in another tank or engine and that sufficient reserve propellant is available to make up for wastage during startup of a faulty engine.

Table I-4-XII shows a list of estimated reliability values for the system components under consideration. These values were estimated from previous experience on various programs and constitute a very conservative estimation when compared with currently advertised values. Data for solid rockets were developed along lines described in Appendix P-A. Data for liquid rockets were based on "most similar" TITAN data, as were the studies in Appendix P-A. Where restart is involved, weighting factors were used as developed in Appendix P-A.

TABLE I-4-XII. ESTIMATED RELIABILITY OF COMPONENTS

Symbols and Assigned Values		P	1-P
P_{B1}	Reliability of Booster 1st Stage	-	-
P_{B2}	Reliability of Booster 2nd Stage	-	-
P_{B3}	Reliability of Booster 3rd Stage	-	-
P_{SO}	Probability that Super-Orbital Abort will not be Required	-	-
P_{E1A}	Reliability of (1)* and (3) Engines for Super-Orbital Abort	0.99025	0.00975
P_{E2A}	Reliability of (2) and (4) Engines for Super-Orbital Abort	0.99025	0.00975
P_{E1M}	Reliability of (1) and (3) Engines for First Mid-course Correction**	0.97250	0.02750
P_{T1M}	Reliability of Tank for First Midcourse Correction	1.0000	-
P_{E2M}	Reliability of (2) and (4) Engines for Second Mid-course Correction*	0.98230	0.0177
P_{T2M}	Reliability of Tank for Second Midcourse Correction	1.0000	-
P_{E1L}	Reliability of (1) and (3) Engines for Lunar Insertion***	0.98770	0.01230
P_{T1L}	Reliability of Tank for Lunar Insertion	1.0000	-
P_{E2L}	Reliability of (2) and (4) Engines for Lunar Exit	0.99473	0.00787
P_{T2L}	Reliability of tank for Lunar Exit	1.0000	-
P_S	Reliability of Re-entry Vehicle Separation Rockets (Solid)	0.9950	0.005
P_{SC}	Reliability of Re-entry Vehicle Spin Control System	0.9923	0.0077
P_A	Reliability of Abort Rockets (Solid)	0.9950	0.005
P_V	Reliability of Attitude Control	1.0000	-
P_M	Reliability of Malfunction Detection System	0.999	0.001

* Numbers in parenthesis refer to engine position on the aft end of the spacecraft.

** Includes 5 starts.

*** Includes 2 starts.

Tankage in the system under consideration is partially redundant as regards safety. However, since the tankage is not wholly redundant, a generous safety margin should be used in the design, especially for H_2 tanks. Reliability of this component will undoubtedly be very high. For the purpose of this study, it will be taken to be 1.00000. The malfunction detection system can be expected to have a reliability of 0.99900. This represents a 50 percent failure-rate reduction over the system under development for Dyna-Soar. By use of these reliability values, the results shown in Tables I-4-12 through I-4-15 were obtained. Attitude control reliability is taken as 0.99900.

The sequencing device for this system would have about the same reliability as the engine sequencer on TITAN Stage II, 0.99900. Malfunction detection and other sequencing is taken as 0.99900.

Table I-4-13 develops the reliability in terms of success in accomplishing the mission with no failures at any phase. The expected reliability is 0.91787 if the booster works properly. Table I-4-13 develops the enhancement due to redundancy possible with the selected configuration. A twofold reduction in failure rate is obtained by the use of redundancy. Probability of completing the mission is approximately 0.95454.

Safety, the most important consideration is developed in Table I-4-14. Since booster reliability is not known this value cannot be exactly evaluated. However, numerical values of safety have been developed for each of the three possible booster stage failures and superorbital abort. Table I-4-XV gives values of safety for various booster reliabilities. Safety after a successful boost phase is 0.97801. The 1966 system would be somewhat improved.

TABLE I-4-XIII. RELIABILITY IN TERMS OF SUCCESS IN ACCOMPLISHING THE MISSION
(Including The Effect Redundancy)

Phase	Description	Probability of Success in Phase	Symbol or Calculated Value For Phase	Cumulative
1	Booster Stage 1	$P_1 = P_{B1}$	P_{B1}	P_{B1}
2	Booster Stage 2	$P_2 = P_{B2}$	P_{B2}	P_{B2}
3	Booster Stage 3	$P_3 = P_{B3}$	P_{B3}	$P_{B1} \cdot P_{B2} \cdot P_{B3}$
4	Super Orbital	$P_4 = P_{SO}$	P_{SO}	$P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
5	First Midcourse Correction	$P_5 = \left[1-(1-P_{E1M})^2\right] \cdot P_{T1M} \cdot P_V$	0.99385	$0.99385\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
6	Lunar Insertion	$P_6 = P_{E1L}\left[1+(1-P_{E1L})P_{E1M}\right]P_{T1L}\ P_V$	0.98215	$0.97608\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
7	Lunar Exit	$P_7 = \left[1-(1-P_{E2L})^2\right]P_{T2L}\ P_V$	0.99390	$0.97015\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
8	Second Midcourse Correction	$P_8 = P_{E2M}\left[1+(1-P_{E2M})P_{E2L}\right]P_{T2M}\ P_V$	0.98442	$0.95504\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
9	Re-entry Vehicle Separation	$P_9 = \left[1-(1-P_S)^2\right]$	0.999975	$0.95502\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$
10	Re-entry Vehicle Spin Control	$P_{10} = \left[1-(1-P_{SC})^2\right]$	0.999410	$0.95454\ P_{B1} \cdot P_{B2} \cdot P_{B3} \cdot P_{SO}$

TABLE I-4-XIV. SAFETY IN TERMS OF SUCCESSFUL RETURNS
(Following Uncorrectable Malfunction at any Phase)

Phase	Mode of Safe Return	[Cumulative Probability of Success Thru Preceding Phase]	[Probability of Single Malfunction This Phase]	[Probability of Success in Return Following Failure in Phase] =	Calculated Value	
		Actual Value Table III				
1-10	Normal Mission	$0.95445\ P_{B_1} P_{B_2} P_{B_3} P_{SO}$			0.95454	$P_{B_1} P_{B_2} P_{B_3} P_{SO}$
1	Booster Stage 1 requires 7 of 8 Abort Rockets, 2 of 2 Large Separation Rockets	1.0000	$1-P_{B_1}$	$P_A{}^7\ (8-7P_A)(P_S)^2\ P_{10}$	0.98171	$(1-P_{B_1})$
2	Booster Stage 2 requires 3 of 4 Abort Rockets, 1 of 2 Separation Rockets	P_{B_1}	$1-P_{B_2}$	$(P_A)^4\ (1-(1-P_S)^2)\ P_{10}$	0.97998	$P_{B_1}\ (1-P_{B_2})$
3	Booster Stage 3	$P_{B_1}\ P_{B_2}$	$1-P_{B_3}$	$(P_A)^2\ (1-(1-P_S)^2)\ P_{10}$	0.98963	$P_{B_1}\ P_{B_2}\ (1-P_{B_3})$
4	Super Orbital Abort use all Engines	$P_{B_1}\ P_{B_2} P_{B_3}$	$1-P_{SO}$	$(P_{E_{1A}})^2 (P_{E_{2A}})^2 (P_{T_1})(P_{T_2}) P_9 P_{10}$	0.96227	$P_{B_1} P_{B_2} P_{B_3} (1-P_{SO})$
5	First Midcourse Correction If (1) and (3) engines fail Return via Cislunar Path	$P_{B_1} P_{B_2} P_{B_3} P_{SO}$	$1-P_5$	$P_M P_8 P_9 P_{10}$	0.00604	$P_{B_1} \cdot P_{B_2} \cdot P_{B_3} \cdot P_{SO}$
	If tank fails: abort and return via cislunar path	$P_{B_1} P_{B_2} P_{B_3} P_{SO}$	$1-P_{TM}$	$P_M P_8 P_9 P_{10}$	0*	
6	Lunar Insertion: If (1) & (3) engines fail return via cislunar path	$0.99385\ P_{B_1} P_{B_2} P_{B_3}$	$1-P_6$	$P_M P_8 P_9 P_{10}$	.01743	$P_{B_1} P_{B_2} P_{B_3}$
	If tank fails, return via cislunar path	$0.99385\ P_{B_1} P_{B_2} P_{B_3}$	$1-P_{T_{1L}}$	$P_M P_8 P_9 P_{10}$	0*	
				TOTAL	0.97801	$P_{B_1} P_{B_2} P_{B_3} P_{SO} +$ $0.98171(1-P_{B_1}) + 0.97998\ P_{B_1}\ (1-P_{B_2}) +$ $0.98963\ P_{B_1}\ P_{B_2}\ (1-P_{B_3}) + 0.96227$ $P_{B_1}\ P_{B_2}\ P_{B_3}\ (1-P_{SO})$

* Since $P_{T_{1M}} = 1$

TABLE I-4-XV. RELIABILITY AND SAFETY SUMMARY CALCULATED FOR VARIOUS VALUES OF BOOSTER AND SUPER ORBITAL RELIABILITY

	Numerical Values For P_{B_1}, P_{B_2}, P_{B_3}, and P_{SO}				
	0.6	0.7	0.8	0.9	0.99
Safety (in Successful Return following failure in any Phase - Table IV)	0.9802	0.9784	0.9790	0.9783	0.9780

4.3 ALTERNATE SYSTEMS STUDIED

4.3.1 Bell Aerosystems Proposed Propulsion System

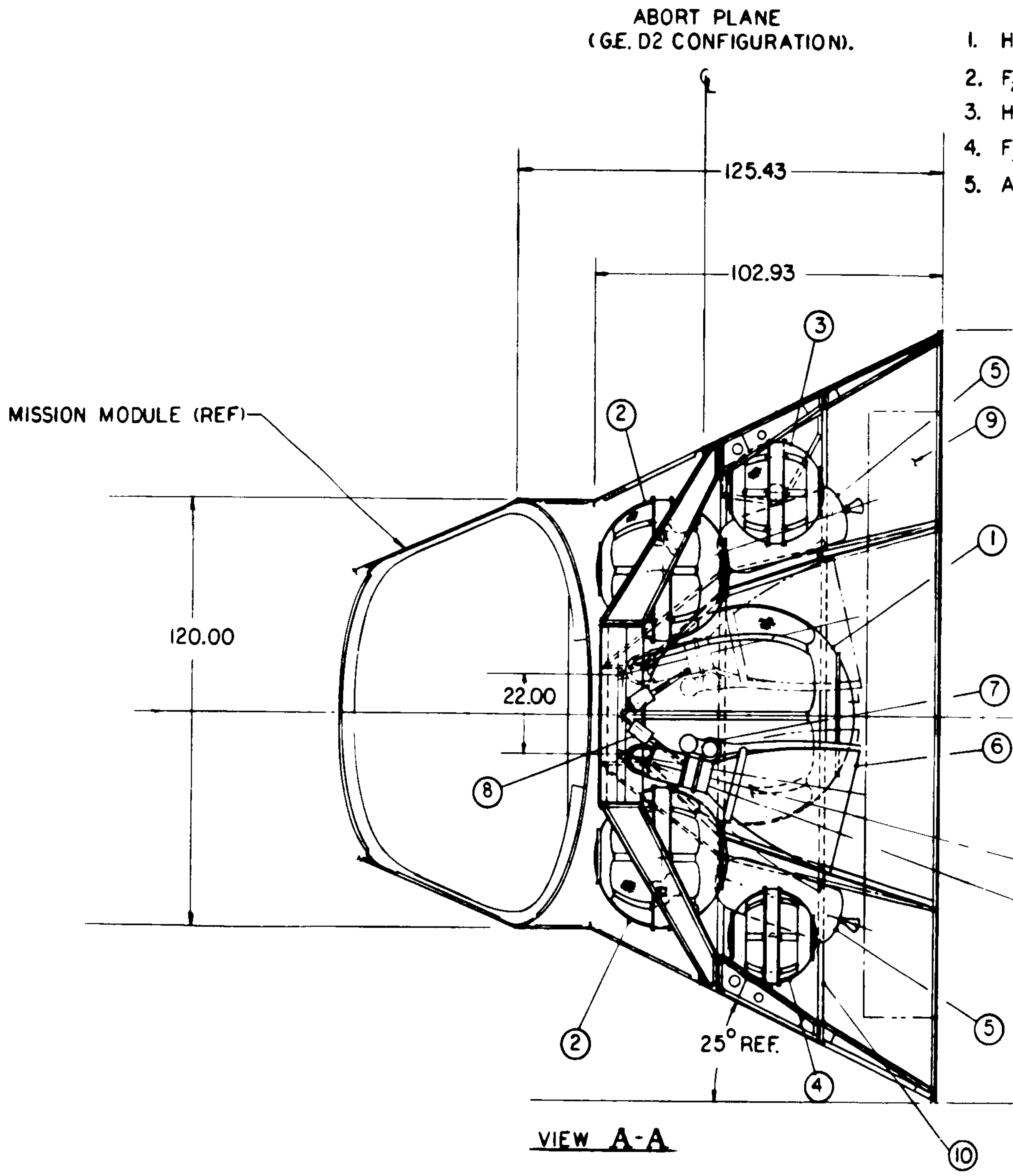
4.3.1.1 SUMMARY

Bell selected the propellant combination liquid fluorine/liquid hydrogen for the main propulsion system coupled with a unique propellant feed system utilizing the better advantages of the turbo pump and pressurization feed. The results of their study are summarized in Appendix P-B and represents an excellent analysis and proposed solution to the APOLLO propulsion. Their engine emphasizes capacity for multi-purpose missions, multiple firings, reliability, and redundancy.

The main propulsion and mission attitude control systems are mounted in a single propulsion module fitting easily within the envelope of the D-2 APOLLO capsule. The total impulse capability of the main propulsion system is approximately 4 million pound seconds. The upper thrust capability is 24,000 lb for superorbital abort and it has a maximum of fifteen restarts of the main engines in space for course correction and lunar orbit and deorbit. Either one or both pump-fed 12,000-lb thrust chambers may be utilized for lunar orbit and deorbit, expending approximately 93 percent of the total usable weight of propellants in four starts. Midcourse corrections are accomplished from a separate helium pressure-fed system to facilitate achievement of the large number of total firings for the maximum mission. Pressure-fed firings are made using the two main engines with thrust decreased to approximately 4000 lb. Bell concludes that the reliability of the pumped/pressure-fed system is essentially equivalent to a pure pressure-fed system.

The proposed system shown in Figure I-4-28 has the further advantage of occupying only a minimum amount of the total available volume, saving perhaps as much as 10 feet of the cylindrical, 10-ft diameter APOLLO mid-section

Detailed technical discussions, schematics, performance, and estimated weight breakdowns, as well as operating sequence and safety and reliability analyses are included in Appendix P-B. In this Appendix, Bell discusses its detailed program plan approach and facilities available for this program implementation.


ABORT PLANE
(G.E. D2 CONFIGURATION).
℄
125.43
102.93
MISSION MODULE (REF)
120.00
22.00
25° REF.
VIEW A-A
1. H
2. F
3. H
4. F
5. A
1
2
3
4
5
6
7
8
9
10

$_2$ LOW PRESSURE TANK — 2 RQ'D.
$_2$ LOW PRESSURE TANK — 4 RQ'D.
$_2$ HIGH PRESSURE TANK — 2 RQ'D.
$_2$ HIGH PRESSURE TANK — 1 RQ'D.
TTITUDE CONTROL & ULLAGE ROCKET PACKAGE — 4 RQ'D.
6. THRUST CHAMBERS
7. TURBINE PUMP
8. GIMBAL ACTUATORS
9. SPACE RESERVED FOR SOLAR ARRAY
10. HEAT SHIELD (DELETED FROM END VIEW)

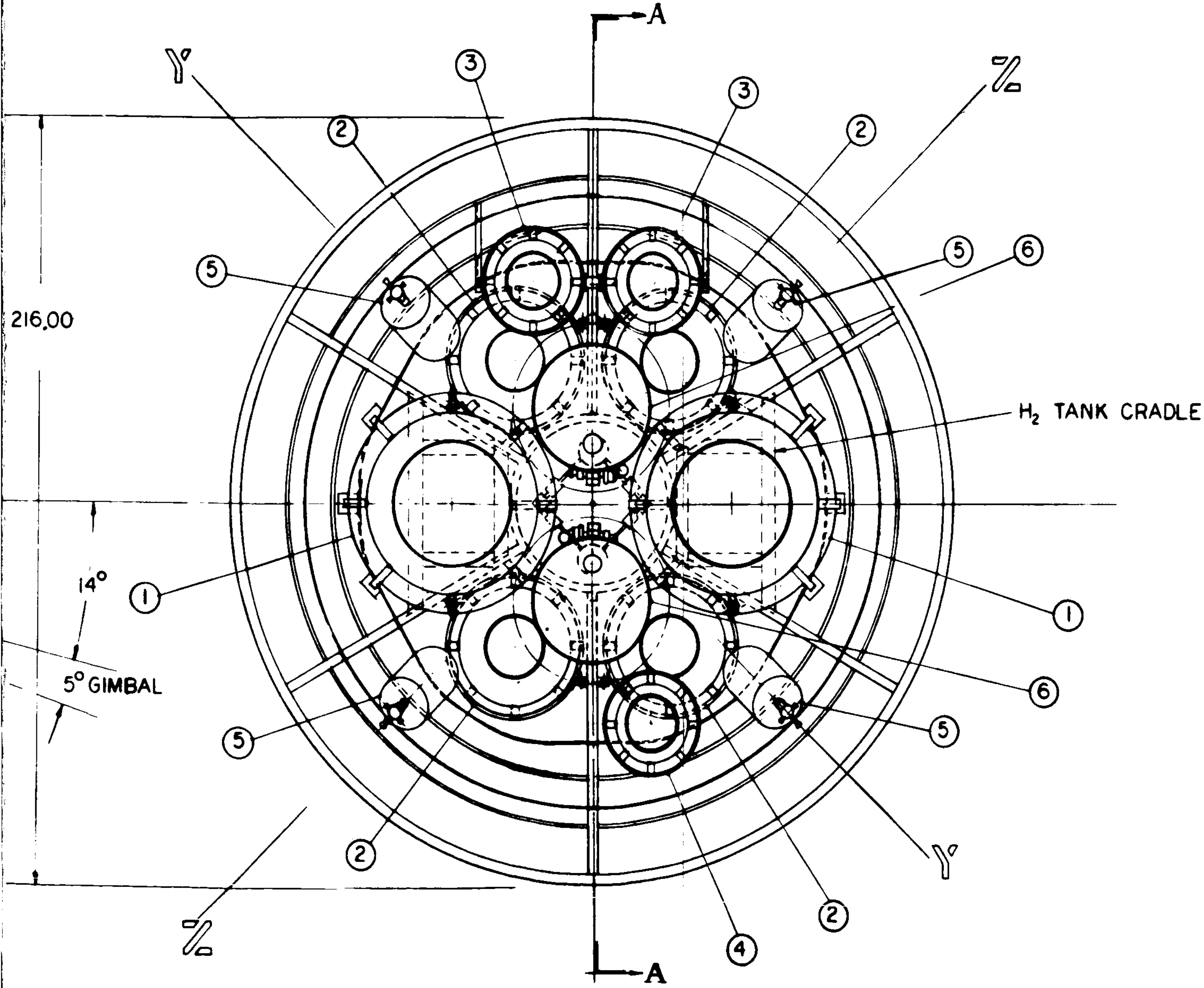


Figure I-4-28. Bell proposed propulsion system

4.3.1.2 DESIGN REVIEW

Bell's proposed use of fluorine hydrogen unquestionably provides an edge in performance and payload over the recommended Aerojet O_2/H_2 system. This edge in performance may be significant for some of the proposed APOLLO missions, but with the resultant payload determined during this study of nearly 8000 lb, both O_2/H_2 and F_2/H_2 exceed the 15,000 lb weight limitation. As described in the parametric study above, there are several alternatives to provide the successful APOLLO mission, but in each case the O_2/H_2 appears adequate. Nevertheless, the Bell system has considerable merit and should be seriously considered for future spacecraft applications, particularly if propulsion volume is limited.

The primary disadvantage of Bell's proposed system is twofold:

(1) System and starting complexity with the proposed combination pumped-fed and pressure-fed system, and

(2) The disadvantages of using liquid fluorine on a manned spacecraft.

Bell Aerosystems and others have been actively working with fluorine for several years and valuable information is now available in storage, handling and testing this propellant. Their report describes some of Bell's detailed experiences and it is their conclusion that fluorine is suitable for manned spacecraft. However, it would appear that there is still substantial work to be done in understanding the storage and handling of liquid fluorine before it reaches the present state of technology of handling oxidizers like liquid oxygen.

A summary of the Bell proposed APOLLO main propulsion system is shown in Table I-4-XVI. These weights were derived for a system capable of providing 7500 feet per second with a 10 percent propellant reserve for a vehicle gross weight of 14,715 lb. These numbers would be adjusted for the particular D-2 vehicle to reflect the increased payload. The tabulated performance is for a single engine which is capable of operating either at 12,000 pounds thrust in the pumped-fed mode or at 4000 pounds thrust as a pressure-fed system.

TABLE I-4-XVI. APOLLO MAIN PROPULSION PERFORMANCE SUMMARY

(1) <u>Requirements</u>	
Vehicle Gross Weight	14,715 Lb
Δ V Total	7,500 Ft/Sec
Δ V Midcourse Corrections	500 Ft/Sec
Δ V Lunar Orbit Exit and Entry	7,000 Ft/Sec
(2) <u>Engine Performance</u>	
a. <u>Pump Fed Engine</u>	
Propellants	Liquid Fluorine/Liquid Hydrogen
Engine Vacuum Thrust, Lbs	12,083
Engine Mixture Ratio, °/F	11.92 ± 1-1/2%
Engine I_{sp}, Nominal, Sec	446.2
Engine I_{sp}, Minimum Observed Guarantee, Sec	443.6
Thrust Chamber Vacuum Thrust, Lbs	12,000
Chamber Pressure, Psia	300
Mixture Ratio, O/F, Thrust Chamber	13
Area Ratio, A_e/A_t	45
I_{sp}, Nominal Thrust Chamber, Sec	448.2
I_{sp}, Minimum Observed Guarantee Thrust Chamber, Sec	446.2
Fuel Pump Discharge Pressure, Psia	465
Oxidizer Pump Discharge Pressure, Psia	400
Turbine Fuel Consumption, Lb/Sec	0.33
Turbine Exhaust Thrust, Lb	83
Exhaust Gas I_{sp}, Sec	250
b. <u>Pressure Fed Engine</u>	
Propellants	Liquid Fluorine/Liquid Hydrogen
Vacuum Thrust, Lbs	3,983
Mixture Ratio, O/F	10
Chamber Pressure, Psia	100
Area Ratio, A_e/A_t	45
I_{sp}, Nominal, Sec	448
I_{sp}, Minimum Observed Guarantee, Sec	445.8

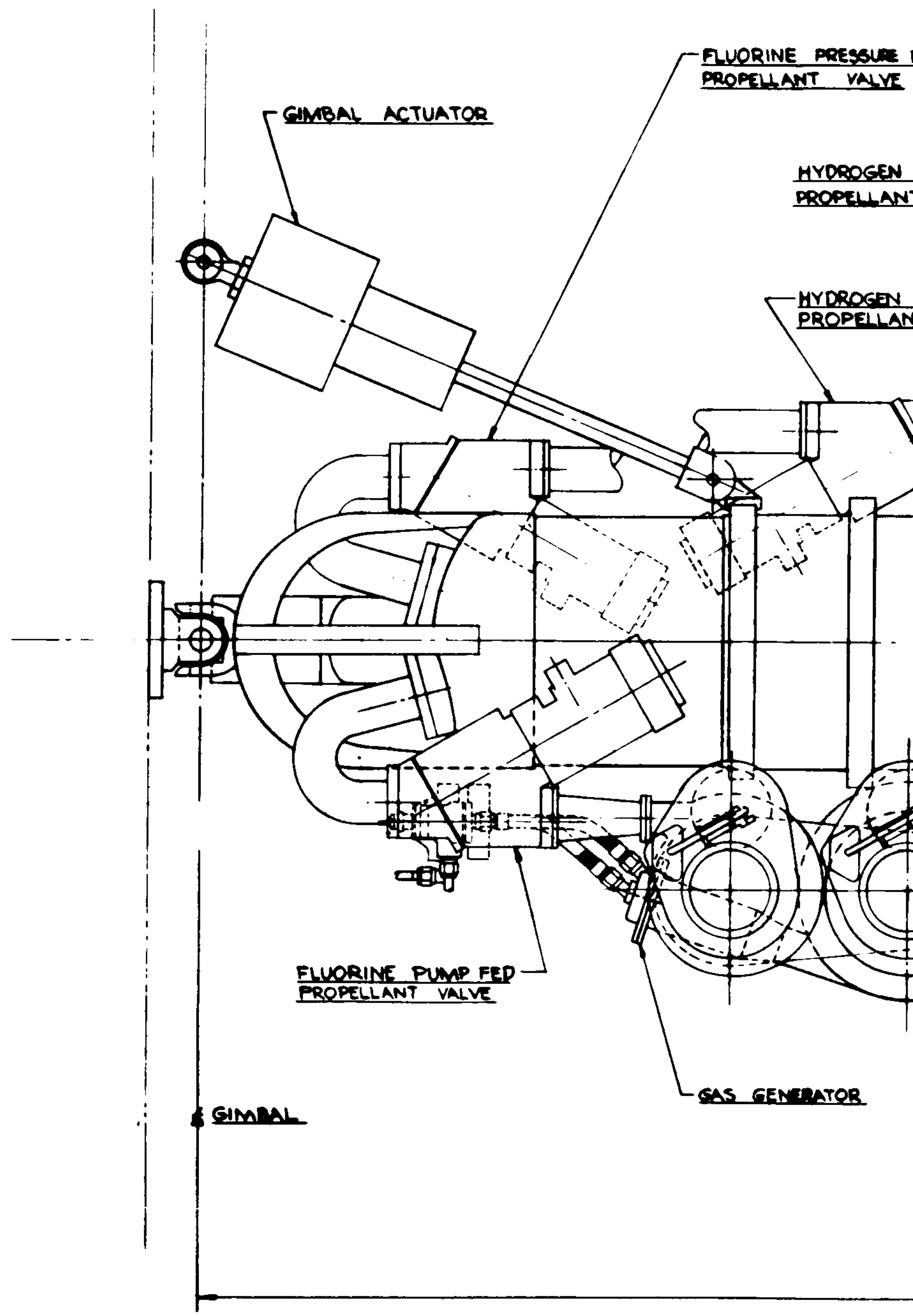
TABLE I-4-XVI. APOLLO MAIN PROPULSION PERFORMANCE SUMMARY (Cont)

APOLLO MAIN PROPULSION WEIGHT SUMMARY (1963 Version)	
Thrust Chamber Assembly	
Including valves, engine mount, turbine pump assembly, gimbal actuators, etc.	555.1
Propellant System	
Including low pressure and high pressure tanks, insulation lines, valves, module structure, tank supports and interconnecting lines	899.0
Pressurization System	
Including helium tanks, tank supports, lines and valves	126.9
Instrumentation Pick-Ups	25.0
Loadable Propellant and Helium	6,831.5
Mission Ullage/Attitude Control System	507.2
Propulsion Module Skin Weight	563.0
TOTAL	9,507.1

The Bell main propulsion system properly emphasizes reliability and safety for this manned spacecraft. Their analysis is shown in Section XII of Appendix P-B and is an excellent piece of work. Bell's reliability analysis evaluates the reliability in terms of mean time between failure. They find that their proposed system appears capable reaching the reliability of 1828 missions between failure. The reliability decrease of the all-pumped system of 14% eliminates the all-pumped system from further consideration for the APOLLO mission. Their analysis shows that the pumped/ pressure fed system is within one percent of the all-pressure fed system.

The complete engine assembly is shown in Figure I-4-29 showing the overall dimensions. The turbo pump assembly is mounted on the thrust chamber from supports about the

BELL AEROSYSTEMS COMPANY



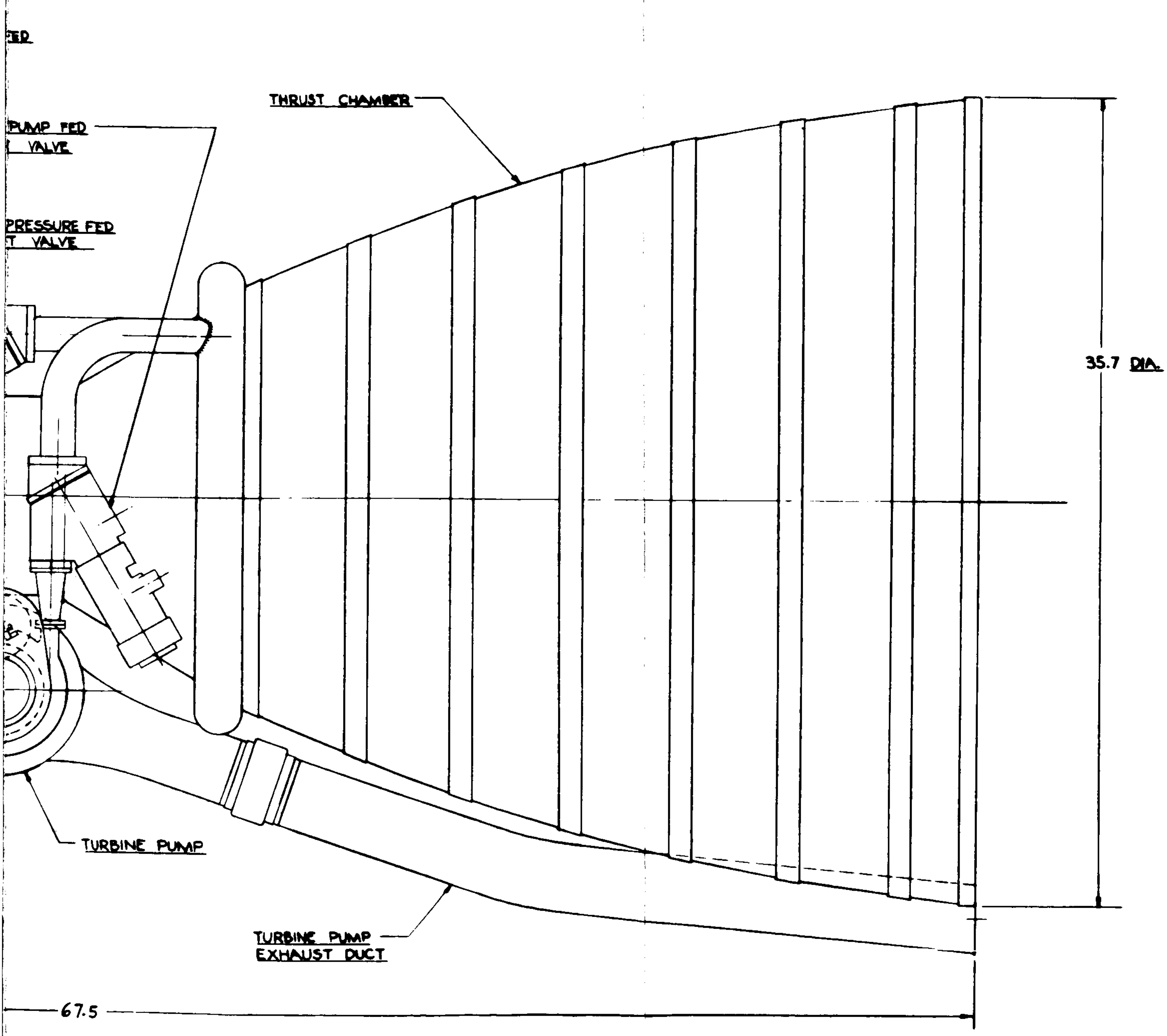
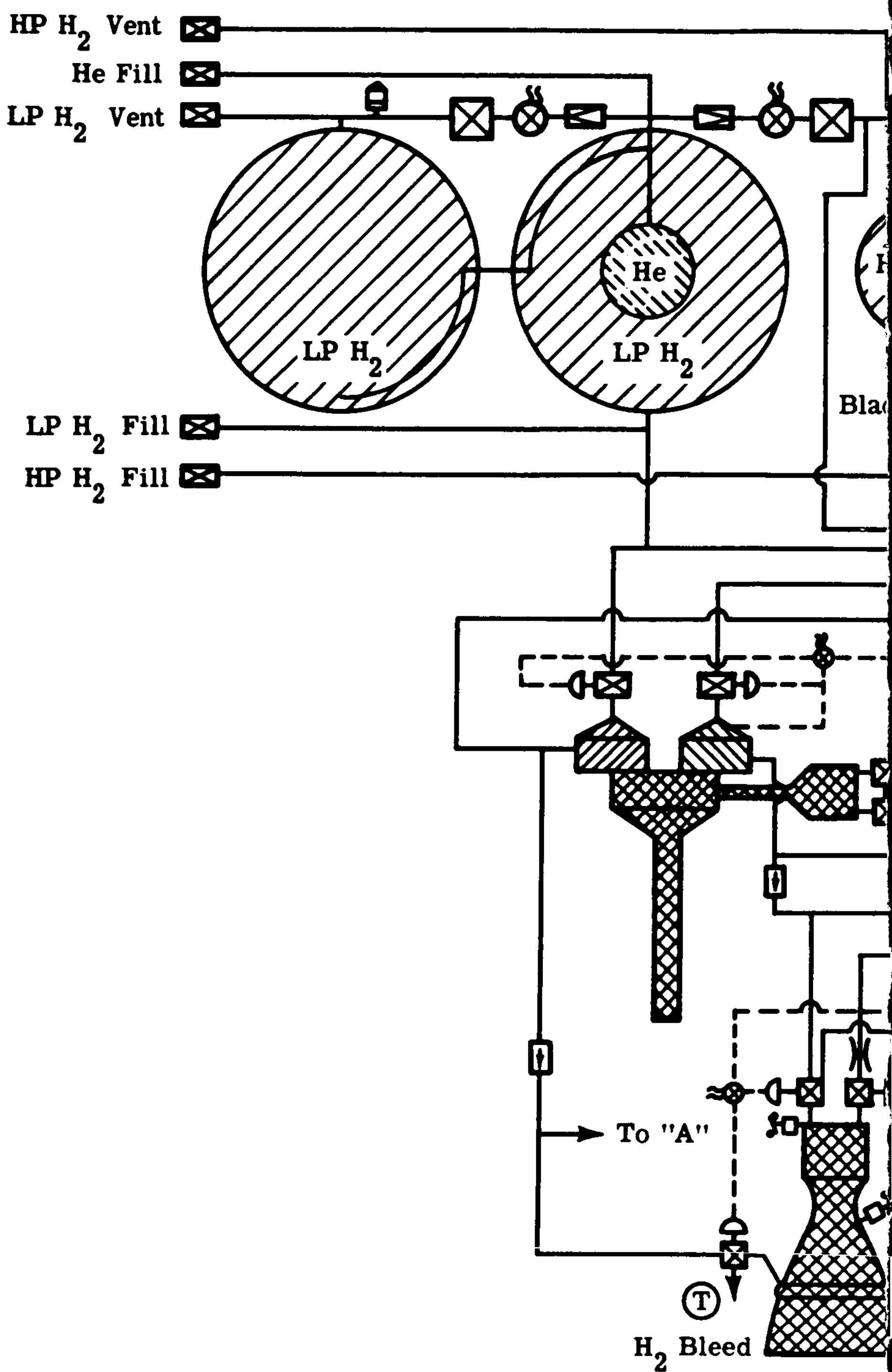


Figure I-4-29. Bell engine assembly drawings throat section. The gas generator, mounted on the turbine inlet manifold, burns fluorine and hydrogen to drive the turbo pump. The complete engine assembly is gimballed to provide thrust vector control of at least 5 degrees in any direction.

The thrust chamber is regeneratively cooled with the hydrogen and operates at pressures of 300 psia during pump fed operation and 100 psia for pressure fed operation. A summary of the thrust chamber characteristics is given in Table I-4-XVII. Additional analysis shows that the thrust chamber could operate at very low chamber pressures (10 to 20 psia) without an injurious temperature rise in the cooling jacket. Such operation at the ultra low chamber pressure would be of particular importance for safety and reliability for lunar deorbit. If there were a multiple failure of both turbo pumped assemblies and the pressurization system, the propellant in the low pressure tanks could be used to accomplish lunar deorbit using the propellant vapor pressures for expulsion and a resultant chamber pressure of 10 to 20 psia. According to Bell's analysis, however, probability of this emergency power requirement is less than one in 2500 missions.

The schematic of the main propulsion system is shown in Figure 1-4-30. The two thrust chambers are turbo-pump fed from low pressure spherical tanks. The chambers can also be operated pressure fed from the high pressure tanks. Thrust levels are 12,000 and 4,000 lb respectively. A helium tank, buried in one of the low pressure H_2 tanks, provides positive suction head pressure requirements of the pumps. This helium is also fed through a second system to the high pressure propellant tanks. The turbo pump assemblies are driven by gas generators which burn F_2 and H_2 in a "boot strap" rise of power. Flow control of the gas generator is accomplished by cavitating venturies. The thrust chamber propellant valves provide bleed flow to cool the propellant pumps prior to starting the gas generator. Temperature sensing elements indicate completion of bleed. After thrust shutdown, bleed ports vent the propellants trapped between the pump inlet valves and the propellant valves. The power acquisition for the pumped fed operation is obtained by ullage rockets incorporated with the attitude control system, augmented by the thrust developed by the gas generator exhaust duct.

The high pressure H_2/F_2 tanks may incorporate bladers for positive expulsion, although additional work needs to be done for use with cryogenic propellants.


HP H_2 Vent
He Fill
LP H_2 Vent
He
LP H_2
LP H_2
Bla
LP H_2 Fill
HP H_2 Fill
To "A"
T
H_2 Bleed

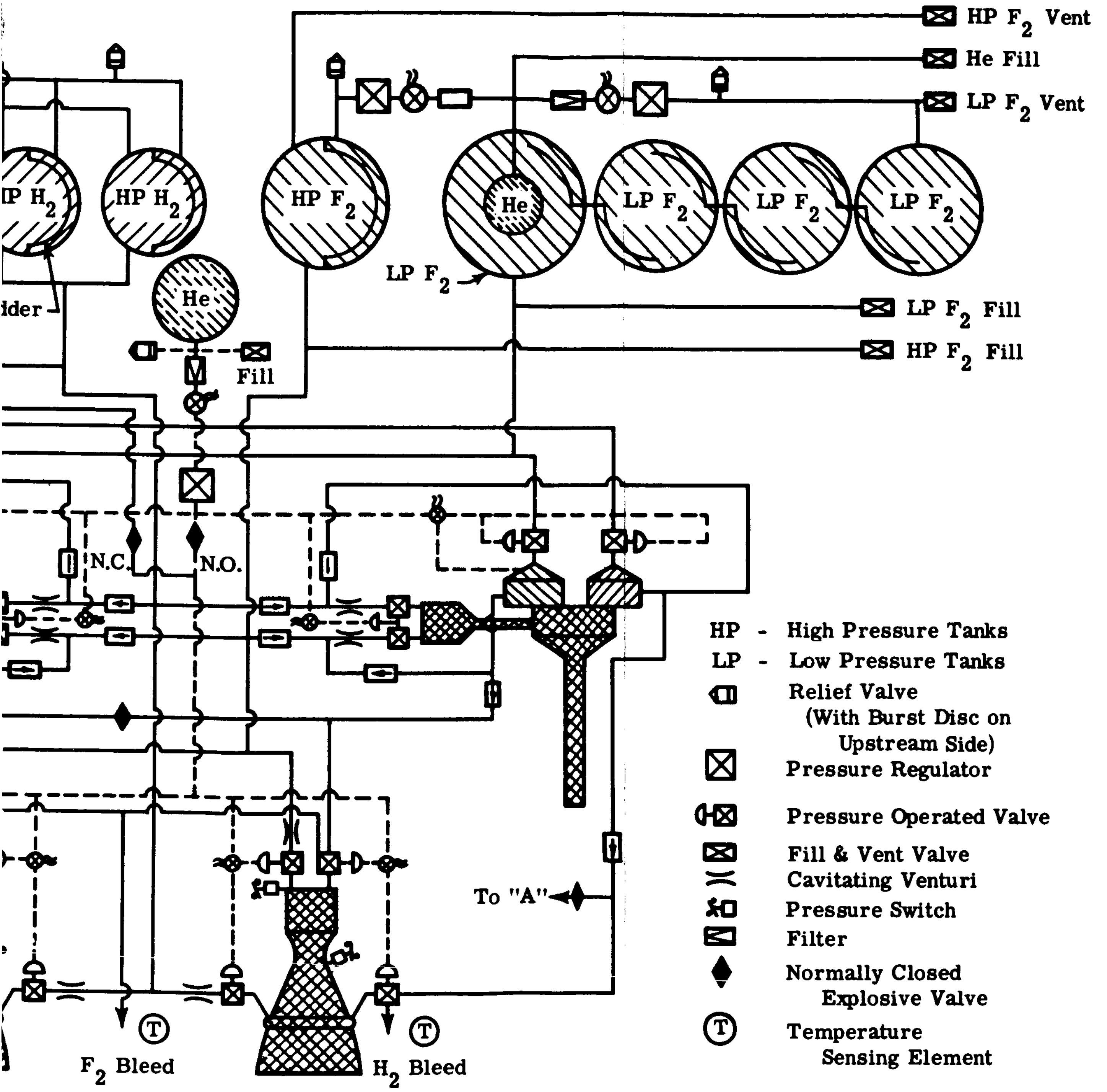


Figure I-4-30. Bell engine schematic

TABLE I-4-XVII. THRUST CHAMBER CHARACTERISTICS

	Rated Conditions	Step Thrust Conditions
Vacuum Thrust, lb	12,000	3,983
Thrust Chamber Pressure, psia	300	100
Mixture Ratio, W_o/W_f	13.0	10.0
Vacuum Specific Impulse, lb sec/lb	446.2*	445.8*
Propellant Flow Rate, lb/sec	26.90	8.92
Regeneratively Cooled Divergent Area Ratio	45	45
Throat Area, sq in	21.545	21.545
Nozzle Exit Diameter, in	35.1	35.1
Oxidizer Feed Pressure, psia	400	111
Fuel Feed Pressure, psia	465	145
Cooling Fluid	Fuel	Fuel

*Minimum observed guarantee vacuum specific impulse; does not allow for 0.7% instrumentation error.

Additional details of the proposed F_2/H_2 system may be found in Appendix P-B.

4.3.2 Reaction Motors Division Proposed APOLLO Powerplant

Reaction Motors has studied the requirements for the NASA APOLLO manned spacecraft in light of their considerable experience with manned rocket engines, including the LR-99 engine currently in use with the X-15 research aircraft. Reaction Motors has studied the APOLLO mission in some detail from the aspects of reliability and quality assurance testing. The report in Appendix P-C includes a particularly interesting basic philosophy for guiding and over-all detail system design.

In summary, Reaction Motors supports the concept that man is the single most important element in the operation of manned spacecraft. Rigorous application of manned qualification and manned safety concepts must constitute a basic philosophy guiding over-all detail system design at every stage of the effort. Further, even with powerplant system reliabilities approaching 100 percent, malfunctions may occur and must be anticipated in the powerplant system or in its environment control. Under such conditions, RMD presents a sound case as to why the powerplant must present no hazard to the vehicle with which it is intimately associated. This, in fact, is a key to the LR-99 engine design concept.

Reaction Motors Division directed their specific design efforts towards achieving a reliable powerplant in a different path than that selected by Bell and by Aerojet. RMD's solution involves a logical growth using storable rocket propellants and powerplants built from existing state-of-the-art technology and available components. They proposed an immediate solution embodying use of current state-of-the-art storable propellants, N_2O_4/MMH. This propulsion system would be available by 1963, and be capable of providing a 24,000-lb thrust with four individual chambers each of 6000 lb thrust.

RMD then proposes the gradual and logical upgrading of vehicle performance and payload through improved propellants, particularly through substitution of oxygen difluoride (OF_2) for nitrogen tetroxide (N_2O_4). They point out that the technology needed to design and develop a helium gas pressurized N_2O_4/MMH system is currently available in the industry. This proposed 1963 system will meet the requirements of an APOLLO circumlunar mission, i.e., it will supply Δ V for space abort of the mission.

RMD feels that research in storage and combustion of their proposed OF_2 during the three year period available for engine system development would promise success of meeting the requirements for the APOLLO lunar orbiting mission by 1966 with storable propellants.

RMD points out, as is indicated in Section 4.1, that the OF_2/MMH propellant combination may be superior in performance to oxygen/hydrogen when compared for overall vehicle performance. RMD states that the storability, handling, and starting of these propellants is simplified because of the hypergolicity thereby eliminating the requirement for an ignition system and simplifying the basic engine.

RMD reports OF_2 to have equivalent performance as the fluorine systems without the compatibility disadvantages of the extremely reactive fluorine. This must be further demonstrated, however.

Use of storable propellants for the main propulsion system facilitates supply to the attitude control system since these propellants can likewise be used for the individual small attitude-control motors. A separate cross-coupled feed system is used for the attitude control system for improved reliability due to the inherent redundancy of the system.

A schematic of the RMD proposed 1963 system is shown on Figure I-4-31. Further details of this engine system and its components are included in Appendix P-C.

4.3.3 Other Existing Propulsion Systems

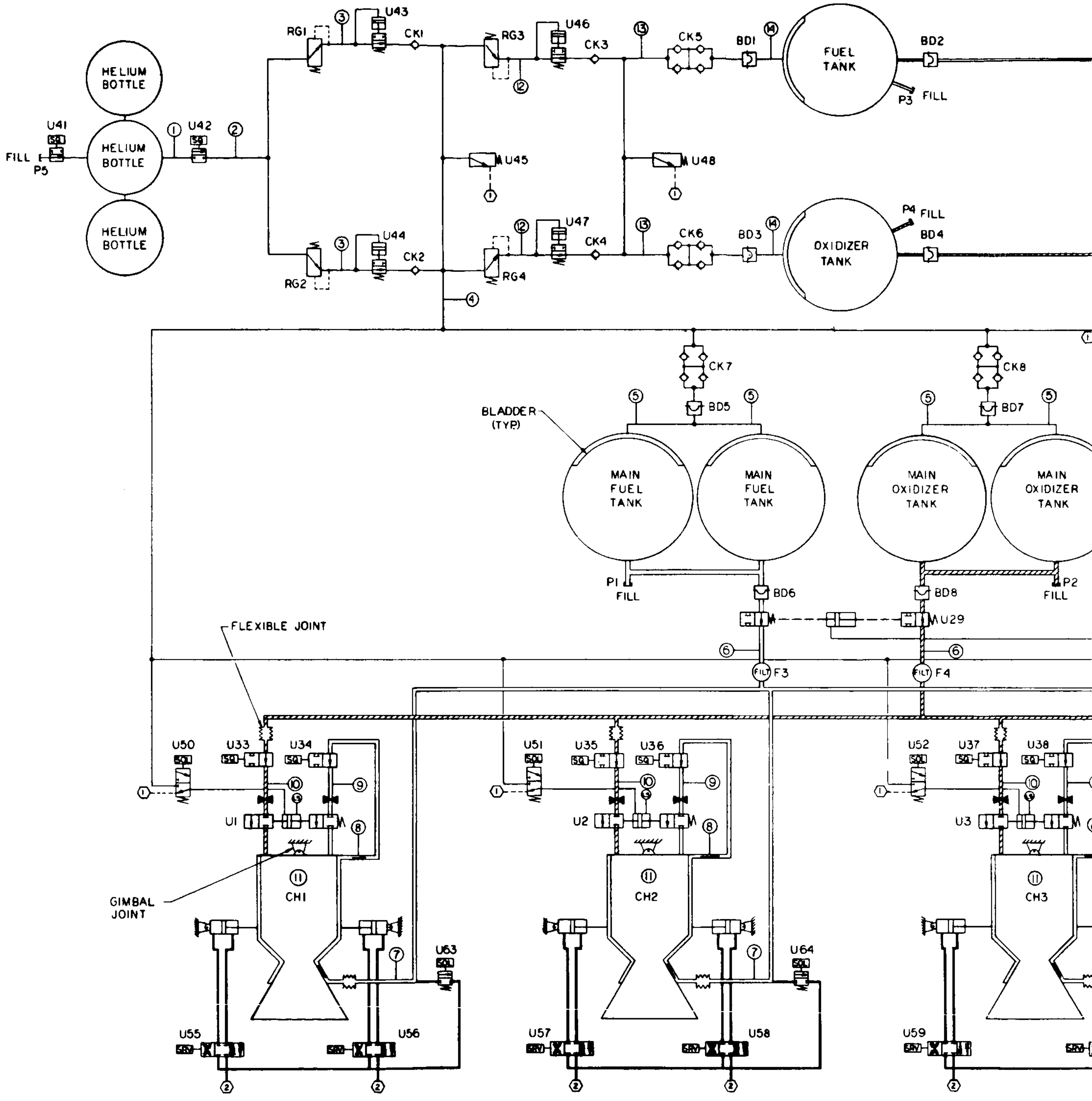
4.3.3.1 CENTAUR ENGINE LR-115

The Centaur engine, LR-115 (LR-10), was briefly examined for this application. Two engines of 15,000 lb thrust each could be used to produce 30,000 lb super-orbital abort thrust. Space maneuvers could be accomplished at a thrust lower than 15,000 lb but it is not clear how much these engines could be throttled.

The LR-115 engines are designed for upper-stage, space vehicle applications and burn liquid H_2/liquid O_2 at nominal O/F ratios of 5 and minimum I_{sp} of 412 sec. Higher I_{sp}, in the vicinity of 420 sec, can probably be obtained in the near future. The LR-115 engine uses a regeneratively-cooled thrust chamber with an expansion ratio of 40:1 and design chamber pressure of 300 psia.

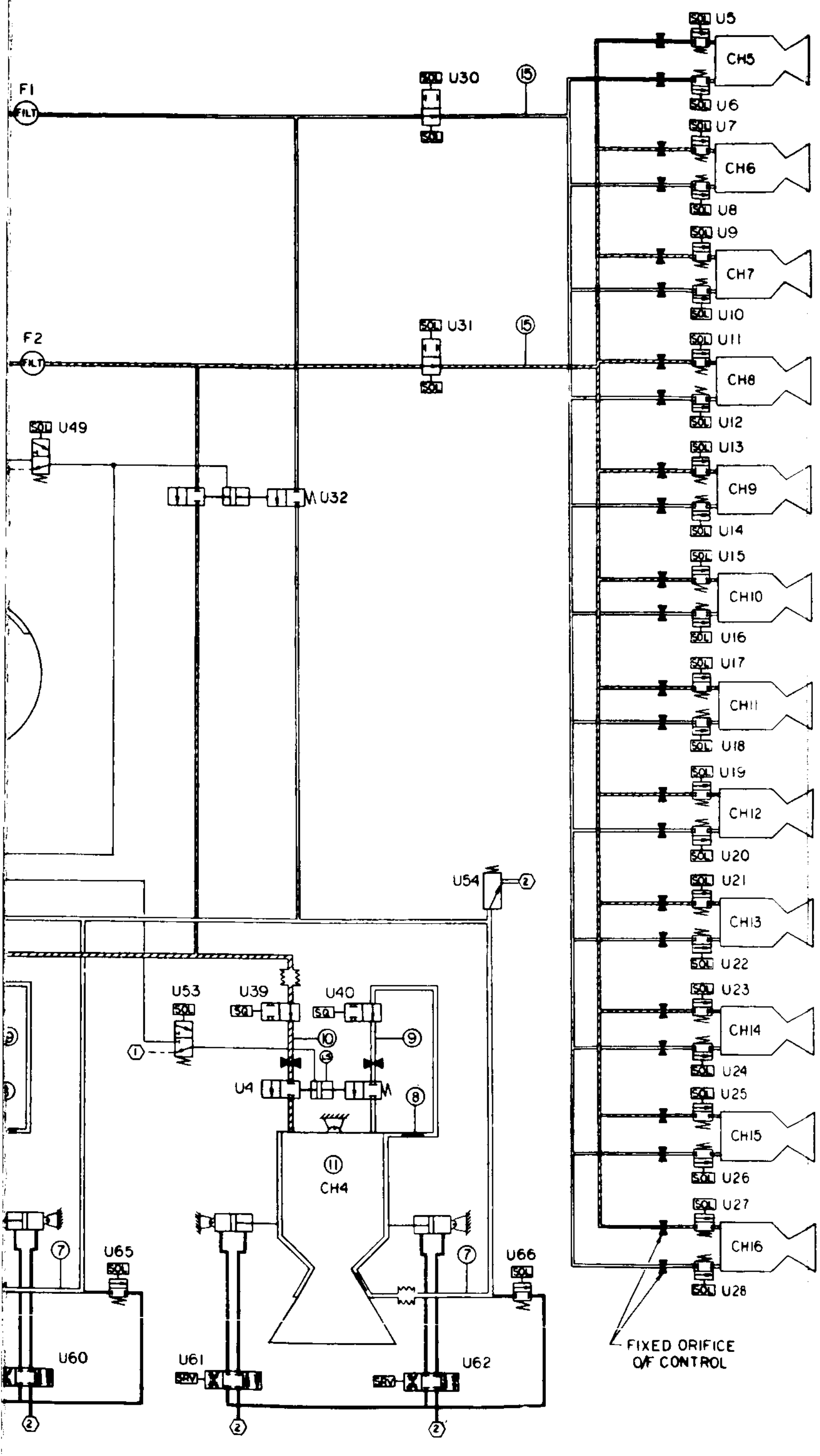
The propellants are pumped by a topping-turbine system which uses the energy of the H_2 from the cooling jacket and, therefore, requires no gas generator. Multiple starts can be accomplished, but with losses of about 60 lb of propellant per engine per start for cool-down. Ignition is by a single electric spark igniter which has not proven to be completely reliable to date but which could be improved by use of multiple spark plugs and possibly by use of O_3F_2 in the O_2 for hypergolicity.

This engine does not appear too desirable for the APOLLO application where numerous space restarts may be required. The inlet conditions to the pump are quite strict



HELIUM BOTTLE
HELIUM BOTTLE
HELIUM BOTTLE
FILL
P5
U41
U42
RG1
RG2
RG3
RG4
U43
U44
U45
U46
U47
U48
CK1
CK2
CK3
CK4
CK5
CK6
CK7
CK8
BD1
BD2
BD3
BD4
BD5
BD6
BD7
BD8
FUEL TANK
P3 FILL
OXIDIZER TANK
P4 FILL
BLADDER (TYP)
MAIN FUEL TANK
MAIN FUEL TANK
MAIN OXIDIZER TANK
MAIN OXIDIZER TANK
P1 FILL
P2 FILL
U29
F3
F4
FLEXIBLE JOINT
GIMBAL JOINT
U50
U51
U52
U33
U34
U35
U36
U37
U38
U1
U2
U3
CH1
CH2
CH3
U63
U64
U55
U56
U57
U58
U59

[redacted]



ITEM	DESCRIPTION
CH1-CH4	MAIN THRUST CHAMBER, GIMBALED
CH5-CH16	ATTITUDE CONTROL THRUST CHAMBER
U1-U4	MAIN PROP. VALVE, PNEU. OPERATED
U5-U28	ATTITUDE CONTL. PROP. VALVE, SOLENOID OPERATED
U29	MAIN PROPELLANT TANK SHUTOFF VALVE, PNEU. OPERATED
U30,U31	ATTITUDE CONTROL SYS. SHUTOFF VALVE, SOLENOID OPER.
U32	CROSS FEED PROPELLANT VALVE, PNEU. OPERATED
U33 U40	MAIN CHAMBER MALF SHUTOFF VALVE, DUAL SQUIB OPER.
U41	HELIUM FILL VALVE, DUAL SQUIB OPERATED
U42	PRESSURIZING VALVE, DUAL SQUIB OPERATED
U43,U44	MAIN HE REGULATOR MALF. SHUTOFF VALVE PNEU. OPER.
U45	MAIN HELIUM PRESSURE RELIEF VALVE
U46,U47	ATTITUDE CONT'L. HE REGULATOR MALF VALVE, PNEU. OPER.
U48	ATTITUDE CONT'L. PRESSURE RELIEF VALVE
U49	MAIN PROP SHUTOFF & CROSSFEED VALVE PILOT VALVE, SOL. OPER.
U50-U53	MAIN PROP. VALVE PILOT VALVE, SOLENOID OPERATED
U55-U62	MAIN THRUST CHAMBER GIMBAL VALVE, SERVO OPERATED
CK1-CK4	CHECK VALVE, SINGLE
CK5-CK8	CHECK VALVE, QUAD.
BD1-BD4	ATTITUDE CONT'L. PROPELLANT TANK BURST DISC
BD5-BD8	MAIN PROPELLANT TANK BURST DISC
F1,F2	ATTITUDE CONT'L PROPELLANT FILTER
F3,F4	MAIN PROPELLANT FILTER
P1,P2	MAIN PROPELLANT TANK FILL PLUG
P3,P4	ATTITUDE CONT'L. PROPELLANT TANK FILL PLUG
P5	HELIUM FILL PLUG
U54	MAIN FUEL MANIFOLD RELIEF VALVE
U63-U66	SERVO FEED VALVE, SOLENOID OPERATED
RG1-RG4	HELIUM PRESSURE REGULATOR

Figure I-4-31. Reaction motor division engine schematic

[redacted]

now but could be moderately reduced in the future with inducers. The present inlet conditions are:

1.	O_2 inlet pressure:	45 psia minimum @	176 R
		130 psia minimum @	209 R
2.	H_2 inlet pressure:	30 psia minimum @	39.2 R
		45 psia minimum @	43.0 R

For APOLLO, considerable care and complexity of propellant storage would be required. Even with booster pumps, like the PESCO pumps used to provide suction head pressures on Centaur, inlet tank pressures must be carefully maintained at sizeable net positive suction head. Further, transient bubbles must be cleared from the propellant lines during starts which would take more than 20 seconds.

Additional information on the Centaur is available in the following references:

(a) P&WA Installation Handbook, RL10 (LR115) Liquid Rocket Engine, dtd. December 1959, Revised 6-1-60

(b) Condensed Summary of Differences Between LR115-P-1 Engine and "Common Centaur and Saturn" LR115 Engine

(c) Specification No. 2222-E, "YLR115-P-1 Engine", Copy No. 87, dated 30 November 1960

(d) P&WA Installation Drawing No. 2024401, Sheets 1 & 2, dated 10-7-60

Because of the numerous incompatible requirements of the LR-115 (LR-10) Centaur engine, this engine is not recommended for the APOLLO mission.

4.3.3.2 NOMAD ENGINE

We have also briefly looked at the NOMAD engine for the APOLLO mission with assistance from Rocketdyne. While Rocketdyne was not selected as one of our study team members, this should not detract from future serious consideration of the NOMAD engine which was designed for manned space applications. Components and technology are available for using this engine on F_2/N_2H_4. According to Rocketdyne, this engine can be readily converted to F_2/H_2.

Numerous tests have been conducted on the components with F_2/N_2H_4, but work has now reached a moratorium until a specific use of the Nomad engine is found. The facilities have been shut down, but Rocketdyne indicates that these facilities can be re-activated to restart work with the Nomad components in a minimum time.

A brief design study was performed by Rocketdyne for both engines, one using F_2/N_2H_4 and the other using F_2/H_2 for the design specifications tabulated in Table I-4-XVIII.

TABLE I-4-XVIII. DESIGN STUDY GROUND RULES

1. Gross stage weight of 15,000 lb
2. Gross stage velocity increment of 7,500 fps
3. F_2/N_2H_4 weight mixture ratio ($\dot{w}o/\dot{w}f$) of 1.6, F_2/H_2 weight mixture ratio of 13.0
4. Two chambers required; zero gimbal angle thrust vectors of chambers pass through intersection of stage centerline and top of propulsion system envelope. Chamber gimbal excursion is ± 4 degrees.
5. Nominal altitude thrust per chamber is 12K at an expansion area ratio of 20 and a (Nozzle stagnation) chamber pressure of 150 psia. (Increased expansion area ratios can be provided by the addition of an uncooled extension to the chamber.) A chamber layout, with expansion area ratios of 20 and 40 defined, is included as Figure I-4-32 for any future design studies.
6. Stage usable propellant weights were calculated from $R = e,^{v/c}$ where R = mass ratio, v = 7500 fps, and c is based on the value of I_s stated in the following paragraph. Usable propellant weight is then Wg (R - 1/R), with Wg equal to 15,000 lb.
7. Nominal altitude I_s for the F_2/N_2H_4 system was conservatively assumed to be 357 seconds which was demonstrated in Nomad testing. A value of 368 seconds

TABLE I-4-XVIII. DESIGN STUDY GROUND RULES (Continued)

was a Nomad design objective and could be obtained were the program to be continued. A value of 437 seconds was assumed for the F_2/H_2 stage. This is 96.5 percent theoretical at a mixture ratio of 13. It was assumed that all stage thrust was coincident with the stage velocity vector.

8. Tank volumes were based on a factor of 1.04 x usable propellant weight to allow for ullage and outage.

9. Basic NOMAD hardware used to the maximum extent possible. This includes the use of the design NOMAD thrust chamber with F_2/H_2 at a mixture ratio of 13.0; this is feasible without any tube modifications.

Comparative parametric study results are shown in Table I-4-XIX showing dry and wet engine weights using nickel (Ni) thrust chambers, and improved aluminum (Al) thrust chamber assemblies. In both cases, tank pressure is about 300 psia; pressurization is by heated helium from storage at 4500 psia and -300 F. A reflux condensor is used atop the F_2 tank to prevent vaporization and consequent venting of F_2 prior to launching.

In summary, the NOMAD engine burning F_2/H_2 from pressurized tanks looks attractive from a payload capacity and performance standpoint. It achieves this advantage, however, using the highly reactive and toxic fluorine, which suggests a long and possibly expensive development program for manned space applications. At this time, it does not appear that the edge in performance over O_2/H_2 compensates for the difficulties in handling, storing, and successfully qualifying the fluorine-hydrogen engine.

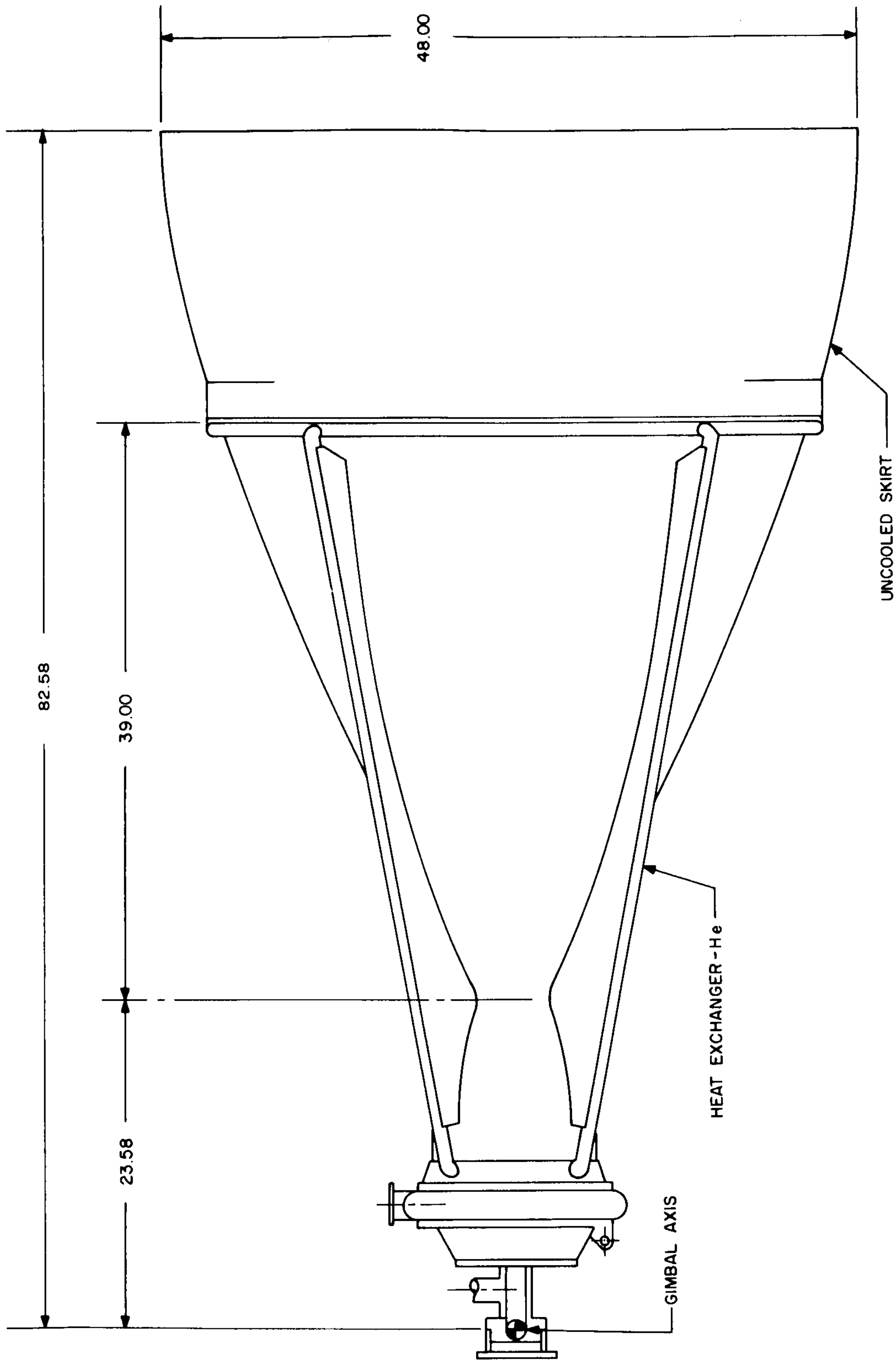


Figure I-4-32. Rocketdyne proposed NOMAD 12k thrust chamber

TABLE I-4-XIX. PARAMETRIC STUDY RESULTS SHOWING APOLLO PROPULSION SYSTEM WEIGHT BREAKDOWN FOR A 15,000-LB VEHICLE, ΔV = 7500 ft/sec

F_2/N_2H_4 System	Weight (Lb) With Ni TCA	Weight (Lb) With Al TCA
Thrust Chambers (2)	380	232
Oxidizer Tank) Fuel Tank)	148	148
Pressure Tank (2)	119	119
Plumbing	63	63
Tank Supports	35	35
Helium	18	18
Dry Engine Weight*	763	615
Propellants	7200	7200
Wet Engine Weight*	7963	7817
F_2/H_2 System		
Thrust Chambers (2)	380	232
Oxidizer Tank & Manifold	167	167
Fuel Tank	156	156
Pressure Tank (2)	119	119
Plumbing (lines, valves, reg., etc.)	70	70
Tank Supports	45	45
Helium	18	18
Dry Engine Weight*	955	807
Propellants	6300	6300
Wet Engine Weight*	7255	7107

*Weights do not include tank insulation, attitude control, structure for engine supports, etc.

4.3.4 Advanced Propulsion System Considerations

4.3.4.1 REVERSE-FLOW ROCKET ENGINES

There are a number of advantages to be garnered from integrating the thrust chamber and tankage with the space vehicle structure. One method of achieving this integration is through use of the reverse-flow nozzles now being studied by several propulsion contractors including the General Electric Co. Rocket Engine Section at Malta Test Station under contract AF 04(611)-6016. One configuration using a reverse-flow nozzle called an "inverted plug" is shown schematically in Figure I-4-33. A more detailed study is described in the RMD Appendix P-C, and has resulted in a design shown in Figure I-4-34.

Basically, the reverse-flow nozzle produces an extremely short chamber by expanding the supersonic flow as a corner expansion around the lower chamber lip to exit axially. Segmenting the chamber into 8 or more units permits adequate redundancy to assure mission safety. Since the chamber is constructed from uncooled, ablative elements,

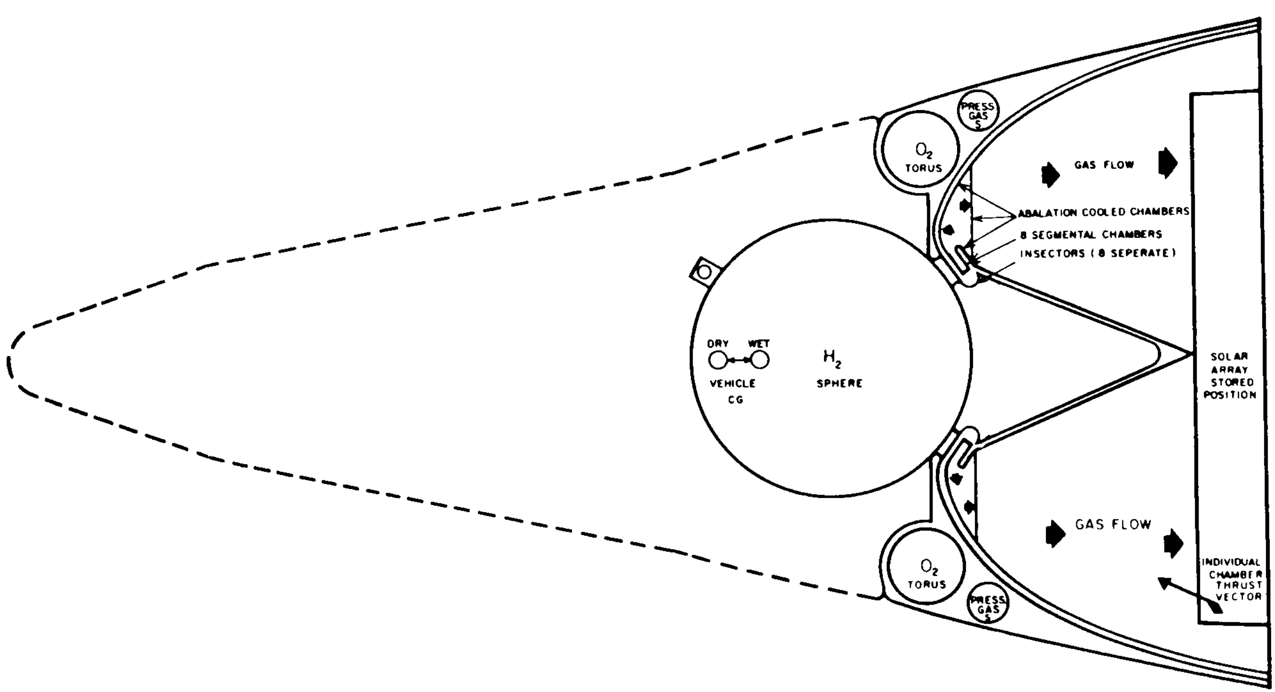
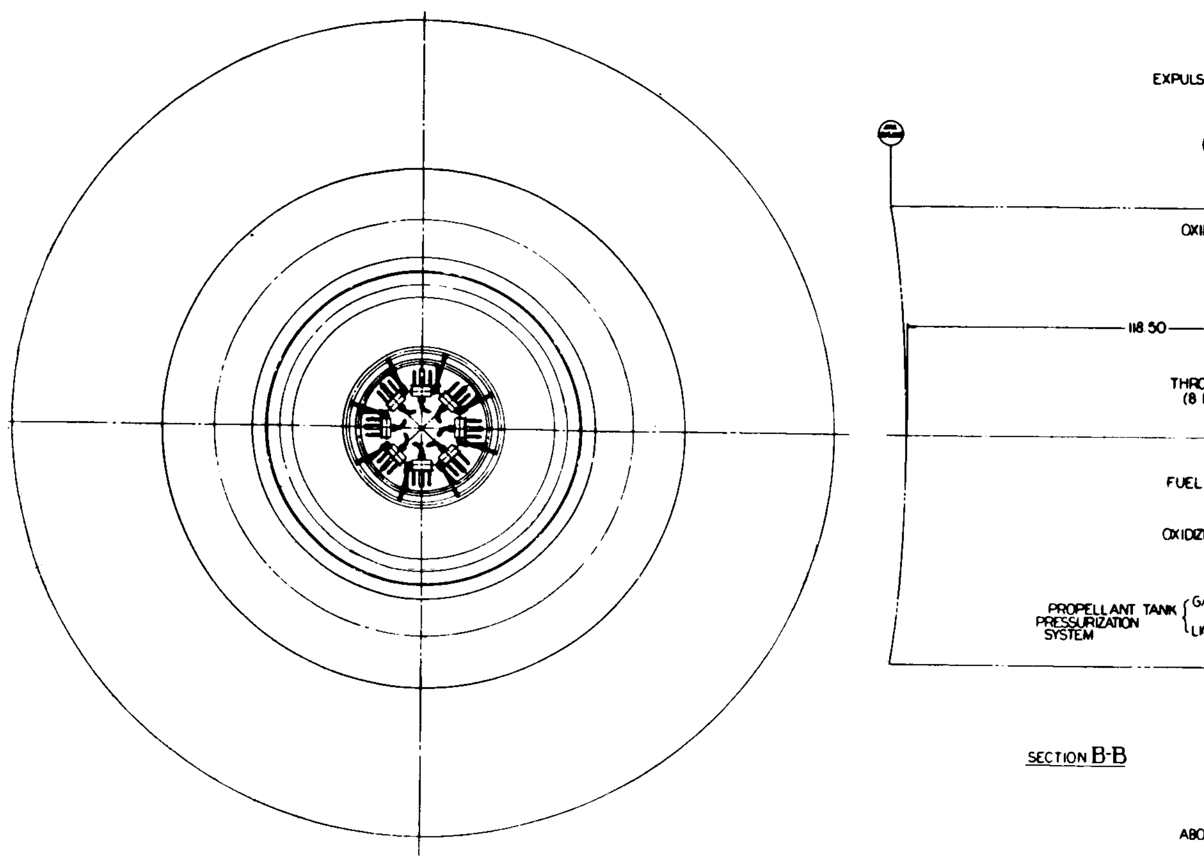


Figure I-4-33. Illustration of reverse flow rocket engine installation the outer walls can double-in-brass for the vehicle structure, as well as transmit the loads directly to the propellant tanks, mission module, and command module.

Since the thrust chamber can now take advantage of the complete available vehicle diameter for rocket gas expansion, the chamber expansion ratio can be raised from 40 to 100 or 200 while still retaining the low chamber pressure. This should result in an increase in specific impulse for the H_2/O_2 system from 427 seconds to 450 seconds, or for the H_2/F_2 system from 440 to 460 seconds, or a similar percent increase with the OF_2/MMH propellants.

The RMD design shown in Figure I-4-34 has a number of attractive features for the APOLLO vehicle. The principal gain is in the simplicity and compactness of this engine. With reduced engine size, the vehicle length and propulsion module volume are substantially reduced with probable attendant savings in weight. In the illustrated design, required vehicle length is reduced from 13-1/2 to less than 4 ft, a saving of 75 percent of the engine compartment length.

Thrust vector capability can be readily achieved by throttling the individual combustors without need for moving the engine physically. Detail description of the RMD engine is presented in Appendix P-C.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FUEL
PROPELLANT TANK PRESSURIZATION SYSTEM
SECTION B-B

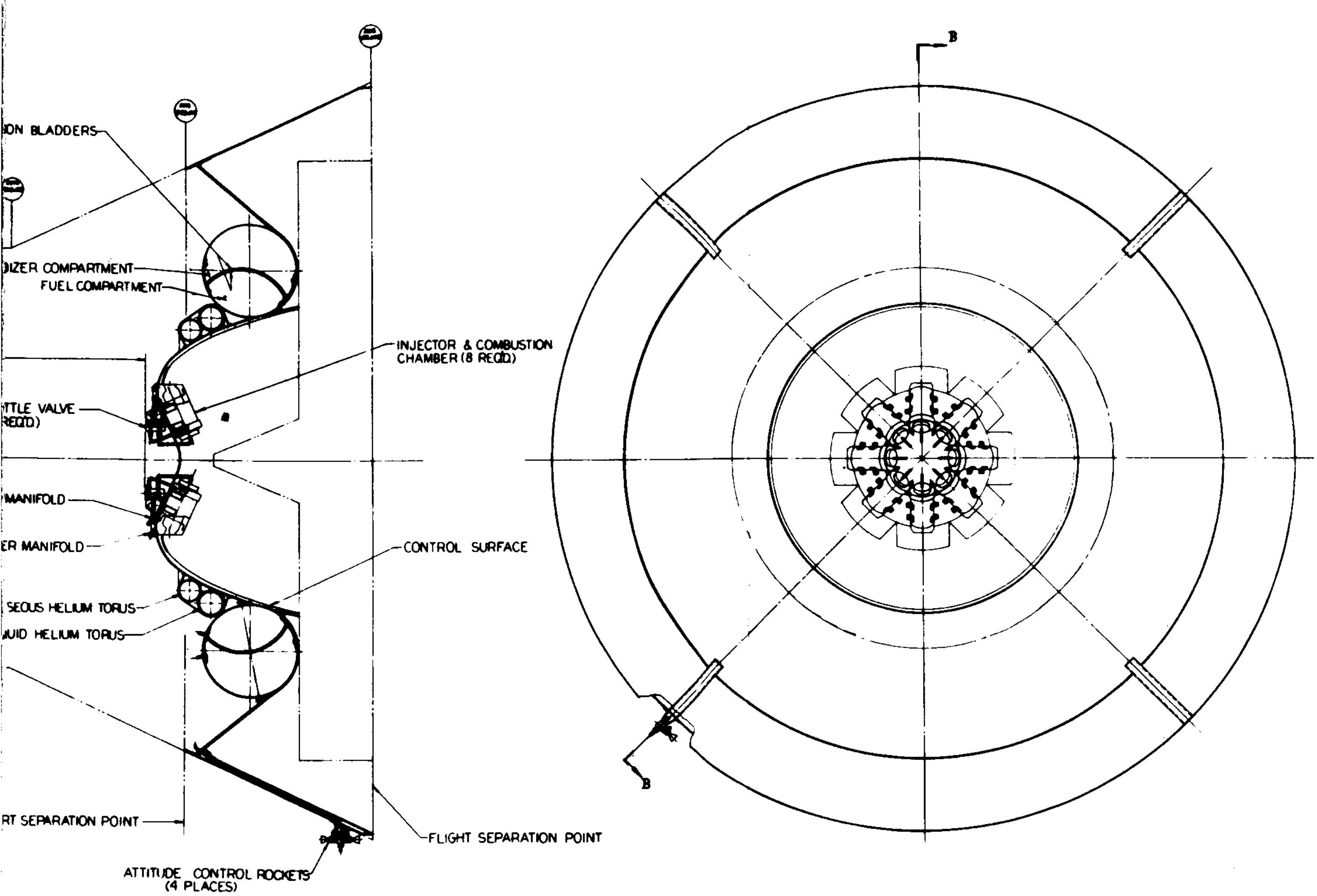

Figure I-4-34. Reaction motors in lunar powerplant, radial chambers installation

5.0 APOLLO Solid Rocket Study-Boost and Separation Rockets

5.1 INTRODUCTION

Maximum reliability is the main criterion used in the design of the solid rocket motors for the APOLLO vehicle. Reliability, in the sense as used above, means the ability of each motor to perform within its design requirements. Thus, "fail safe" is not an adequate definition, and the definition must be extended a step further. Each motor will be designed so that successful ignition and specified performance will be delivered <u>only</u> on command. This is predicated by the fact that the Saturn will not be "man rated" and, consequently, it is expected that the abort system will be needed at some period during the APOLLO program.

Early in this program, the General Electric Company determined the abort requirements during boost to escape velocity of the APOLLO vehicle, could most satisfactorily be met by solid motors as typified by their high thrust, short duration, short reaction time, and high reliability aspects. It also became obvious that the thrust required of the solid motors will be greatest for on-the-pad, lift-off, and during first-stage burning. The thrust requirements continually decrease during the second and third-stage burning. The weight of the on-the-pad abort propulsion is quite significant and considerable vehicle weight saving can be achieved by discarding the excessive units during second and third-stage burning. It therefore appears prudent to use multiple abort units. This further enhances the chances of safe abort, for should one unit fail to ignite, the remaining units insure a reasonable chance of survival. This is true especially in regions of the trajectory where the solid rocket capability is in excess of that which is required. The units will be jettisoned from the APOLLO vehicle as they are no longer required. Abort motor weight now becomes a less critical factor and consequently was given a lower order priority as compared with reliability. Aerojet-General Corporation and Thiokol Chemical Corporation designs reflect these guidelines.

5.2 STATEMENT OF INPUT REQUIREMENTS

The solid motor designs are in accordance with the enclosed list of specifications and were supplied as input data to the two subcontractors. Consequently, the designs in the attached appendices are structured about these input specifications. The basis and justification of these specifications are discussed more fully in the Abort section of this report.

5.2.1 Input, Solid Rockets (Design Parameters)

1. <u>ABORT</u> (Ballistic Vehicle)

1. Initial Abort g's (In the direction of thrust)	20
2. Burning time, sec.	1.0 for 6 units 2.0 for 2 units or 2.0 for 8 units
3. Aborted weight (Exclusive of Abort Propulsion)	7,000 (1963)
4. Number of Abort Units	8
5. Units dropped at end of First Stage	4
6. Units dropped at end of Second Stage	2
7. Abort Rocket Angle (mounting), degrees	25
8. Net Thrust Vector through abort c.g., degrees	15 off vertical

<u>SOLID LARGE SEPARATION ROCKETS</u> (Ballistic Vehicle)

1. Number of units	4
2. Burning time, sec.	0.75 (Approximately)
3. Unit thrust (each motor), lb	11,000 (Approximately)

<u>SOLID SMALL SEPARATION ROCKETS</u> (Ballistic Vehicle)

1. Number of units	4 (forward) 4 (aft)
2. Unit thrust (each motor), lb	625 (Approximately)
3. Burning time, sec.	1.0 (Approximately)

2. <u>ABORT</u> (Glide Vehicle)

1. Initial Abort g's (In direction of thrust)	15
2. Burning time, sec.	1.9

3.	Aborted weight (Exclusive of Abort Propulsion), lb	6000 (1963) 5500 (1966)
4.	Number of Abort Units	6
5.	Units dropped at end of First Stage	4
6.	Units dropped at end of Second Stage	None
7.	Motor nozzle cant angle from centerline of vehicle, degrees	20

Changes in the abort criteria, which have occurred after completion of the sub-contractors' solid motor designs, have not been reflected in their analysis. These changes which apply only to the ballistic vehicle abort rockets are as listed below.

	ITEM	CHANGE FROM	TO
1.	Initial Abort g's (In direction of thrust)	20	15
2.	Burning time, seconds	2.0(8 units)	2.5(8 units)
3.	Aborted weight (Exclusive of Abort Propulsion), Pounds	7000(1963)	7280(1963)
7.	Abort Rocket Angle (mounting), degrees	25	30
8.	Net thrust vector through abort c.g., degrees	15	20

Each subcontractor has stated that changes within the same order of magnitude will not appreciably affect their system design, philosophy, or budgetary cost estimates. Thus, the major portion of the enclosed reports are applicable, and the technical design parameters are truly representative of what can be supplied in the solid area.

5.3 SOLID ROCKET SYSTEM SELECTION

The selection of the solid motor designs used in this report is made on the basis of the technical contents of the reports submitted to the General Electric Company by the propulsion subcontractors, namely, the Aerojet-General Corporation and Thiokol Chemical Corporation. These reports are attached in their entirety as appendices to this report.

Our evaluation of the two approaches shows little basic difference between the two designs. Thiokol has, however, indicated a greater depth of coverage in such areas as program plan, effects of hard vacuum and related tests, heat transfer analysis, and cockpit display parameters. Consequently, the salient portions of the Thiokol solid motor designs are included herein and are shown on the General Electric configuration drawings.

5.4 SOLID ROCKET PROPELLANT SELECTION

Thiokol Chemical Corporation examined three major propellant systems for possible application to the APOLLO solid motor designs. They are, as given in Section F of the Thiokol Report, EP41-61 (Appendix P-D): polysulfide, polyurethane, and polybutadiene acrylic acid.

The PBAA propellant was chosen because of its high burning rate, high specific impulse, and excellent physical properties and because a vast amount of knowledge exists from proven designs. This propellant is used in all designs with the exception of one which utilizes an existing engine with slight modifications. Some pertinent properties of the PBAA propellants are listed below.

Type		TP-H-3041A
Burning rate @ 1000 psia		= 0.50 in/sec
Specific Impulse,	Pc = 1000 psia Pa = 14.7 psia Pe = 14.7 psia	= 247 lb_f-sec/lb_m
Temperature Sensitivity		= 0.12%/°F
Burning rate Sensitivity		= 0.072%/°F
Characteristic Exhaust Velocity		= 5114 ft/sec
Theoretical Flame Temperature (Chamber)		= 3390°K

Roughly three and one-half million pounds of PBAA propellants have been processed to date by Thiokol. Many existing propulsion systems including both stages of Pershing and the first stage of Minuteman utilize this propellant. Vacuum performance has been demonstrated at simulated altitudes in excess of 100,000 ft. Further, a significant amount of data on the effects of soft vacuum on aging is available. Thiokol has proposed a test program for the small separation motors which is designed to determine the effects of hard vacuum aging, radiation, and other vacuum effects on the propellant.

Polysulfide propellant is used in the large separation motors because it represents the present propellant used in the existing Thiokol TE 146 Cherokee. This motor is recommended, along with an alternate choice using PBAA propellants, and is discussed more fully under the description of the Ballistic Vehicle large separation motors. The Polysulfide propellant is characterized by a specific impulse of 220 seconds referenced at 1000 psia and sea level. Further details are presented in Appendix P-D.

Higher specific impulse propellants were also considered. Comparatively little experience has been accumulated with these propellants and using such propellants will lower our confidence level. Further, as will be shown later, the effects of jettisoning the abort rockets, produce a weight penalty changed against the APOLLO vehicle of one pound for every 3.5 lb of abort rocket weight. Thus, saving 3.5 lb in the abort rocket weight will produce a weight saving in the APOLLO vehicle of one pound. Because of the lower confidence which will exist in the design and because of the small potential weight savings, very high specific impulse propellants are not recommended for the APOLLO abort motors.

The eight small separation rockets used with the ballistic vehicle are carried through the complete mission. Their unit weight, however, is only 3.36 lb and thus, the conclusions reached above also apply.

5.5 BALLISTIC VEHICLE SOLID PROPULSION STUDY RESULTS

5.5.1 Abort Rockets

Two abort rocket designs are presented, each consisting of eight units with the same thrust but with different burning times, i.e., one second and two seconds. The two-second units designated EPD-310 are used herein. These motors are mounted on the vehicle as shown on Figure I-5-1. Four of the eight motors will be jettisoned at first-stage burnout, two at second-stage burnout, and the remaining two at burnout of the third stage of the Saturn booster. Weight penalties which must be charged against the APOLLO vehicle are 2 percent of the weight jettisoned at first-stage burnout and 12 percent of the weight jettisoned at burnout of the second stage of the Saturn booster. All abort propulsion weight carried to third stage burnout must be charged to the APOLLO vehicle weight. Thus, if the jettisoned unit weight is W_j, the abort weight

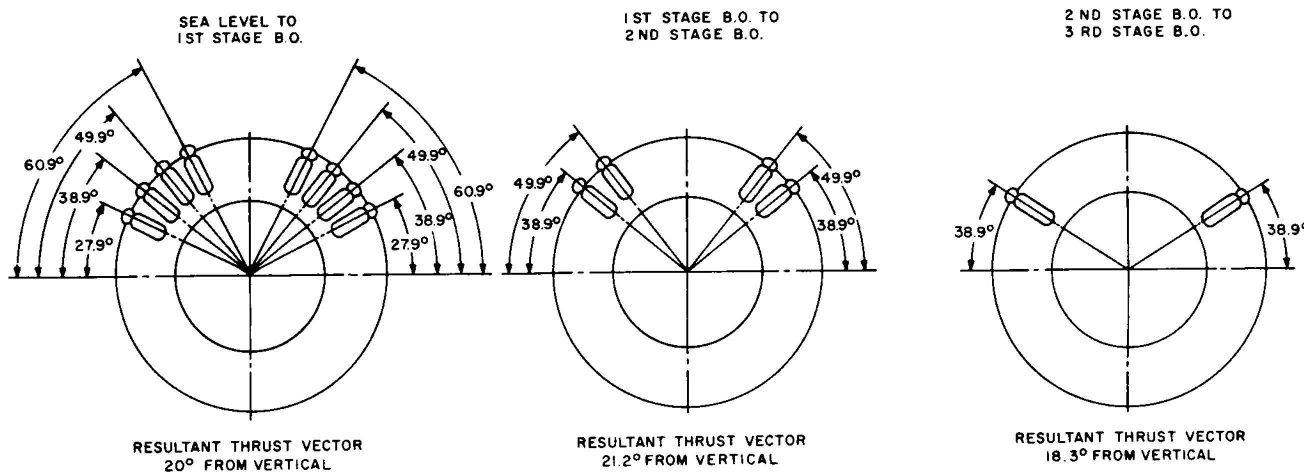

Figure I-5-1. Mounting location of 8 EPD-310 abort motors, top view

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penalty is $(0.02)(4)\, w_j + (0.12)(2)\, w_j + 2\, w_j = 2.32\, w_j$. The on-the-pad abort motor weight is 8 w_j plus the weight of attachments which remain on the vehicle. The Thiokol design gives this weight as 4lb per engine or 32 lb total. Since this weight is small compared with the jettisoned weight (roughly 1800 lb), the weight penalty charged against the APOLLO vehicle weight is 2.32/8 = 0.29 times on-the-pad weight. Thus, each pound charged to the vehicle weight represents roughly 3.5 lb of abort rocket weight.

The reliability of each individual motor is predicted by Thiokol as 0.999. Thus, the probability that all eight of the eight motors will operate successfully is 0.992. The porbability that seven of the eight or eight of eight will operate is 0.99997. This value neglects any effects of one failure on the remaining seven motors. Consequently, the probability that more than one noncatastrophic engine failure will occur is extremely remote. Figure I-5-2 shows the thrust-to-weight ratio as a function of time with all engines firing and with seven of the eight engines firing. With one engine out, the thrust-to-weight ratio is 18 as compared with the nominal 20.

A comparison of the weight savings available by using multiple abort units as opposed to one large unit has been made. The Thiokol data show a total abort motor phase attachments and fittings weight of 1829.6 lb. It is estimated that one unit can be designed to produce the same total thrust and burning time for a weight of 1550 lb. This single motor will have to be carried to escape if abort propulsion capability is provided throughout boost. Its total weight of 1550 lb must be charged against APOLLO vehicle weight. The effective weight penalty of the eight abort rockets is 552 lb as compared with their total weight of over 1800 lb. Consequently, using eight motors in lieu of one motor for abort will have a net weight savings on the order of 1000 lb.

Figure I-4-1 of Section 4.1 shows that this 1000 lb will result in a reduction of 440 ft/sec of the velocity capability of the D-2 vehicle.

An additional consideration affecting the choice of multiple units versus one unit is reliability. Assume that the basic reliability of each motor (one of multiple units or a single unit) is 0.999. This is comparable with the unit reliability for the Thiokol

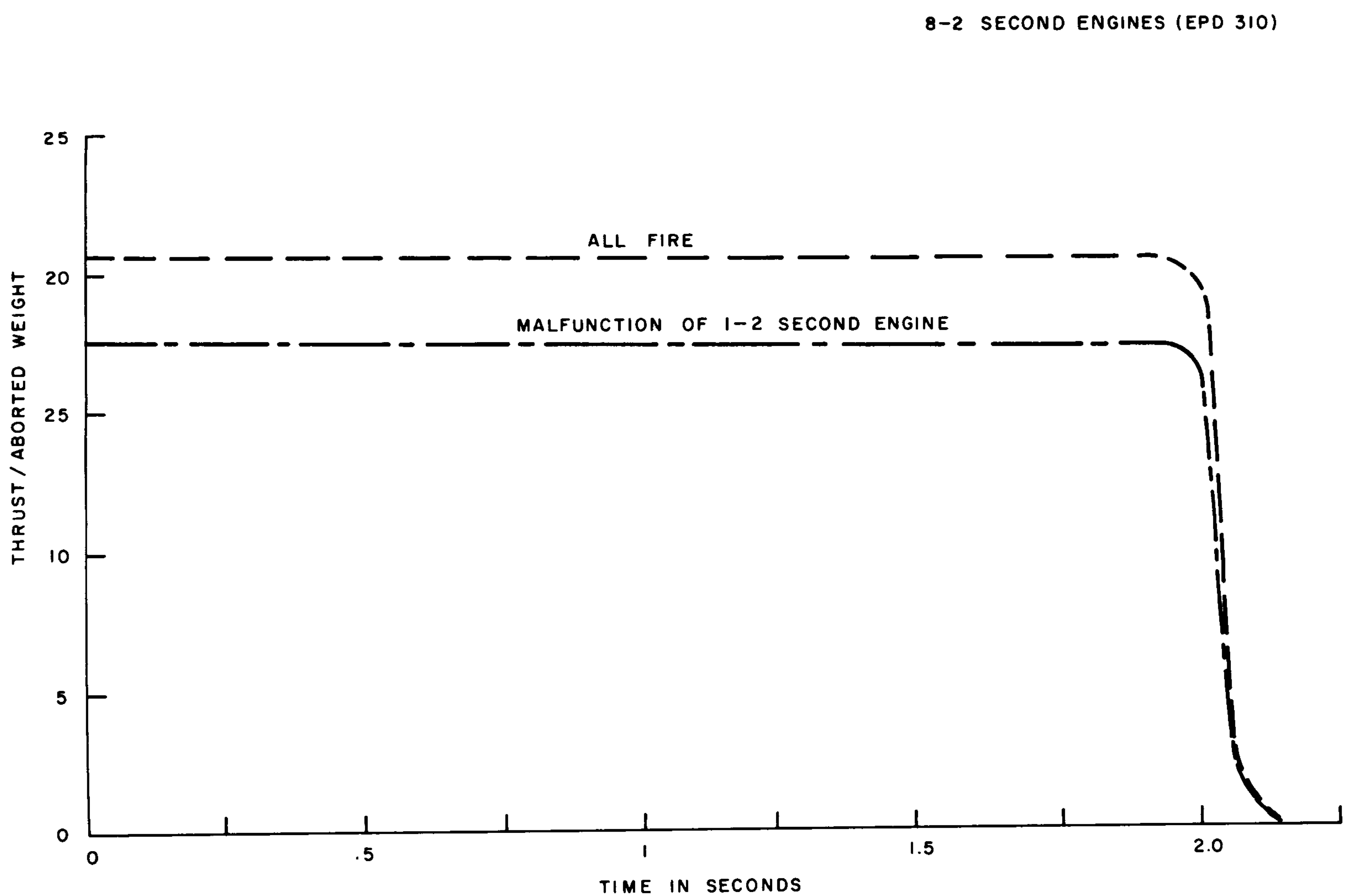


Figure I-5-2. Thrust to weight ratio vs. time (on the pad abort) multiple unit design motors. Using the data as stated earlier, the probability that seven of the eight of the abort propulsion thrust will exist is 0.99997 for the multiple units, if one unit fails. This may be compared with no abort capability if the single unit fails.

Consequently, because of their weight and reliability advantages, multiple abort units are recommended for the APOLLO vehicle.

Another aspect of the abort propulsion is its independence of total vehicle weight. The abort sequence which uses the solid motors, involves aborting only the re-entry vehicle and the command module. This is discussed more fully in the abort section of the report. The main on-board APOLLO propulsion remains with the third stage of the Saturn booster. Consequently, weight of the solid abort rockets is only half of that necessary to abort the complete APOLLO vehicle. The total abort motor weight is therefore a function only of the re-entry vehicle weight and the command module weight and is independent of the weight of the main on-board propulsion system.

5.5.2 Large Separation Rockets

Four Thiokol EPD-316A large separation rockets are proposed to separate the mission module from the re-entry vehicle during high drag on the pad abort. A total thrust of roughly 44,000 lb for approximately 0.75 sec is specified. These specifications are substantiated in the abort section of the report.

This capability is needed only during the high-drag regions. The drag decreases to nearly zero during burning of the Saturn second stage. Thus, as discussed earlier, it is advisable to jettison these units as soon as they are no longer required. This should occur no later than third-stage ignition in which case the penalty charged to the APOLLO vehicle weight will be, as discussed earlier, 12 percent of the weight which is jettisoned.

The four proposed units, EPD-316A are slightly modified versions of the existing Thiokol TE-146 Cherokee solid rocket engine. The modification consists of a change in ignition location, i.e., from the nozzle end to the head-end of the motor. The total weight of these units including attachments, safe and aim, thrust adapter, etc., is 342.4 lb. The weight penalty to carry these units to second stage burnout is 55.2 lb, 16 lb of which consist of unjettisoned rails.

The reliability of these motors will be at least as high as the motors which must be developed for abort.

The motor burning time is given as 1.12 sec which is greater than the minimum requirement of 0.75 sec. Modification of this engine or building a new engine with a 0.75-sec burning time will produce a total effective weight saving of roughly 6 lb. (12 per cent of 50 lb.) This major modification or new design appears unwarranted in light of the low weight saving and, therefore, is not recommended.

An alternate approach is also proposed for the large separation rockets. It is comprised of using two of the EPD-310 abort motors instead of the four EPD-316A units. Their total initial thrust is 47,400 lb for 2 sec. As such, they have far greater capability than is required but, at the same time, they will eliminate one solid motor test and qualification program. The cost of the EPD-316A proof test and qualification program for a 15 unit-proof test and 50 unit qualification program (MIL-R-25534A(I-X) is $470,000. This is rather a low figure when compared with the total APOLLO onboard propulsion cost. The total weight of the two EPD-310 installation is 457.4 lb. The penalty to carry through thrust-stage burnout is 61.9 lb. The net weight penalty for using two EPD-310 engines instead of four EPD-316A engines is therefore 6.7 lb.

Another aspect which must be considered is reliability. Consider a basic unit reliability of 0.9990. The following breakdown is shown:

<u>EPD-316A - Four Units</u>

Probability that 4 of 4 will be successful	- 0.996
Probability that 3 of 4 or 4 of 4 will be successful	- 0.999988
Probability that 2 of 4 or 3 of 4 or 4 of 4 will be successful	- 0.999994

<u>EPD-310 - Two Units</u>

Probability that 2 of 2 will be successful	- 0.998
Probability that 1 of 2 or 2 of 2 will be successful	- 0.999998

Thus, the probability that half the required thrust will be available is roughly the same. However, the four-unit configuration has a high probability that three quarters of the thrust will be available. This possibility does not exist with the two-unit configuration.

The use of four EPD-316A motors is favored because of the higher total thrust capability if one unit should fail. The thrust is tailored to the maximum dynamic pressure abort condition. The required thrust decreases on either side of this point. Basically, this means that should one unit fail with either configuration, the four-unit configuration will be satisfactory over a wider range of the applicable portion of the trajectory.

5.5.3 Small Separation Rockets

Eight Thiokol EPD-312 rockets are provided to separate the re-entry vehicle from the on-board propulsion package and from the mission module prior to re-entry into the earth's atmosphere. Four of these units are mounted with their nozzles facing forward and are used to separate the on-board propulsion. The remaining four units, having aft facing nozzles, separate the mission module from the re-entry vehicle.

These motors must be carried with the APOLLO vehicle throughout the mission and so are constructed of a fiberglass case and an integral plastic nozzle and expansion cone. PBAA type propellant is also used in these motors.

The weight of these eight motors is 26.9 lb. Each safe and arm mechanism weighs 2 lb for a total of 16 lb for the eight motors. If S & A units are used, the total weight for the eight-motor installations is 42.9 lb.

Four units are recommended for each application because of the higher total-thrust capability with one engine failure. Much of the same reasoning given in the previous discussion under large separation rockets is applicable here.

5.5.4 Ballistic Vehicle Solid Rocket Summary

A summary of the Ballistic Vehicle Solid Motor Design details is presented in Table I-5-I. Additional details are presented in the Appendix P-D.

TABLE I-5-I. SUMMARY OF THIOKOL SOLID MOTOR DATA FOR APOLLO BALLISTIC VEHICLE

Item	Abort Motors	Large Separation Motors		Small Separation Motors
		Recommended	Alternate	
Number of motors	8	4	2	8
Thiokol Designation	EPD-310	EPD-316A	EPD-310	EPD-312
Propellant	TPH-3041A	TPL-3014	TPH-3041A	TPH-3041A
Diameter, in.	12.2	5.02	12.2	3.26
Motor Length, in.	56.3	64.1	56.3	17.1
Nozzle Expansion Ratio	9.3	5.7	9.3	36.0
Average Thrust S. L. @ 60 F, lb	21,700	10,240	21,700	642 (Vac)
Burning Time @ 60 F, sec	1.96	1.12	1.96	1.00
Specific Impulse, S. L. @ 60 F, sec	250.4	230	250.4	292 (Vac)
Total Impulse S. L. @ 60 F, 16 sec	44,820	12,000	44,820	642 (Vac)
Propellant Weight, lb	179.6	52.2	179.0	2.21
Motor Weight, lb	215.7	72.6	215.7	3.36
Motor Assembly Weight, lb	228.7	85.6	228.7	5.36
Total System Weight, lb	1829.6	342.4	457.4	42.9
Total Weight after Jettisoning, lb	32	16	8	---
Penalty Charged to APOLLO Vehicle, lb	552	55.2	61.9	42.9

Data presented by Aerojet-General are quite similar in most respects. Total impulses for the abort motors are somewhat larger than those given by Thiokol. The reason for this is a higher impulse in the tail-off of the Aerojet design. Consequently, the slightly higher Aerojet abort motor weight.

The changes that are mentioned earlier for the ballistic vehicle abort rockets have been evaluated by General Electric. Our estimates show that only a minor change will occur in the weight of the abort motor assembly, i.e., from 228.7 to 238.5 lb per unit. The average unit thrust level will be decreased from 21,700 to 17,800 lb. The penalty for the dropping sequence given earlier is changed from 552 to 568 lb.

5.6 GLIDE VEHICLE SOLID ABORT MOTOR PROPULSION STUDY RESULTS

Solid propellant rocket motors are used on the APOLLO Glide Vehicle for abort only. Six Thiokol EPD-311 motors are required, four of which are jettisoned at first-stage burnout. The remaining two are jettisoned at third-stage burnout. The penalty factors established earlier show that this jettison sequence has a penalty of (0.2) (4) $w_j + 2 w_j = 2.08 w_j$ where w_j is the jettisoned unit weight. The on the pad abort motor weight is 6 w_j plus 24 lb of attachments which remain on the vehicle. Assuming this attachment weight is small in comparison with the jettisoned weight (roughly 1100 lb), the weight penalty charged against the APOLLO glide vehicle is $2.08/6 = 0.39$ times the jettisoned weight. Consequently, only 1 lb of penalty is incurred for each 2.5 lb of abort motor weight.

These motors are very similar in design to the EPD-310 abort motors used in the APOLLO ballistic vehicle. The major difference is the use of a canted nozzle and a slightly shorter propellant grain. The canted nozzle is dictated by the mounting of the motors on the vehicle and the shorter grain is a result of the lower thrust required from the glide vehicle abort motors.

Thiokol also predicts a unit reliability of these motors of 0.999. The following systems reliabilities then apply:

Probability that 6 of 6 will be successful	-0.994
Probability that 5 of 6 or 6 of 6 will be successful	-0.99997

The same conclusions reached earlier, with respect to multiple units, also apply here.

The following table presents a summary of the Thiokol glide vehicle abort motor design.

TABLE I-5-II. SUMMARY OF THIOKOL SOLID ABORT MOTOR DATA FOR APOLLO GLIDE VEHICLE

Number of Motors	6
Thiokol Designation	EPD-311
Propellant	TPH-3041A
Diameter, in.	12.2
Length, to nozzle at centerline of plane exit, in.	47.7
Nozzle Expansion Ratio	11.6
Average Thrust, S. L., @ 60 F, lb	17,230
Burning Time, @ 60 F, sec	1.97
Specific Impulse, S. L., @ 60 F, sec	250.5
Total Impulse, S. L., @ 60 F, lb-sec	35,820
Propellant Weight, lb	143.0
Motor Weight, lb	175.2
Motor Assembly Weight, lb	188.2
Total System Weight, lb	1129.2
Total Weight after Jettisoning, lb	24
Penalty Charged to APOLLO Vehicle, lb	407.1

5.7 SOLID MOTOR PROGRAM PLAN

Below is a brief summary of Thiokol's proposal program plan for the APOLLO solid motor development and qualification programs. Additional details are presented in Appendix P-D.

The program plan is presented for the solid motors for an APOLLO ballistic vehicle and for an APOLLO glide vehicle. Basically, a 12-month development program and a 6-month qualification program are presented for each vehicle design. Thiokol can deliver any or all of the motors 18 months from program commencement. Further, if needed, this program can be compressed in time to a period of roughly 12 months. The following table shows the number of tests which are proposed.

TABLE I-5-III. PROPOSED TESTS

Item	Proposed Tests (Number)			
	EPD-310	EPD-311	EPD-312	EPD-316A
Development				
Ignition	15	15	30	15
Motor	30	30	40	15
Ejection System	30	30	--	--
Qualification	50	50	50	50

NOTE: EPD-310, 312, and EPD-316A are for the ballistic vehicle and EPD-311 is for the glide vehicle.

Fifty qualification tests are believed by Thiokol to be sufficient to adequately qualify the motors. The program schedules are based upon this number of tests. Figure I-5-3 is a typical program plan for the 2-sec ballistic vehicle abort rockets.

The types of tests proposed for the abort and large separation motors and for the small separation motors are shown respectively on Figure I-5-4 and I-5-5.

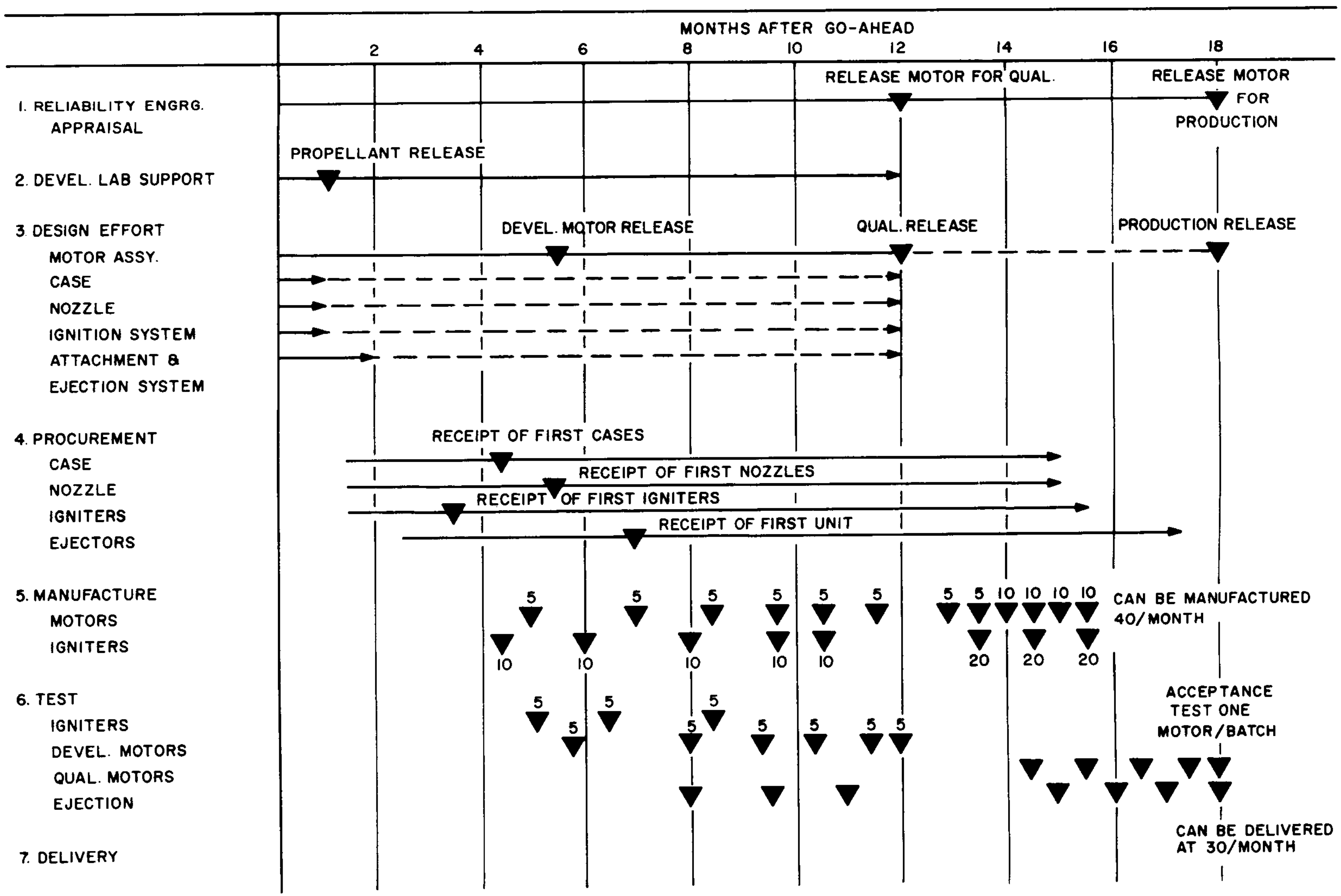


Figure I-5-3. Program plan two second abort motor

1. Case Hydroburst
2. Case Hydrotest
3. Igniter Hydroburst
4. Attachment Fitting Loading
5. Motor Assembly Hydrotest with Thrust Force
6. Igniter Hot Tests in Evacuated Vessel
7. Motor Hot Tests with Chamber Evacuated at Ignition

    a. Temperature Cycle
    b. Hot and Cold
    c. Vibration
    d. Shock
    e. Humidity
    f. Rain and Salt Spray
    g. Sequential

8. Vacuum Tunnel (AEDC)
9. Jettison and Thrustor (Loaded Motors)

Figure I-5-4. Types of tests proposed abort and large separation motors

1. Case Hydroburst
2. Case Hydrotest (All Cases)
3. Igniter Hydroburst
4. Motor Assembly Hydrotest with Thrust Force
5. Attachment Fitting Loading
6. Igniter Hot Tests in Evacuated Vessel
7. Motor Hot Tests with Chamber Evacuated at Ignition

    a. Temperature Cycle
    b. Hot and Cold
    c. Long Term, High Vacuum Aging
    d. Vibration
    e. Shock
    f. Humidity
    g. Rain and Salt Spray
    h. Radiation Exposure
    i. Sequential

8. Vacuum Tunnel (AEDC)

Figure I-5-5. Types of tests proposed small separation motor

6.0 Attitude Control

6.1 REQUIREMENTS

Attitude control is required for APOLLO to maintain vehicle orientation in space and is used in conjunction with the Navigational System. Consequently, the attitude control system requirements are determined primarily by the method of navigation. Calculations based on the navigational system employed in APOLLO have yielded a torque requirement of 110 lb-ft in pitch and yaw, and 140 lb-ft in roll. Torque of this magnitude requires a reaction system, such as a flywheel, to supply the torque for the necessary length of time. However, the flywheel would weigh in excess of 100 lb. With one flywheel per axis, the total weight would exceed 300 lb. With the present vehicle configuration, a reaction system utilizing couples could have a separation of approximately 18 ft, reducing the necessary thrust to 6 lb per engine for pitch and yaw, and 8 lb per engine for roll (two engines per couple). Calculations for navigational maneuvers and limit cycle operation (given in Appendix M of Volume III) have yielded an impluse requirement of 50,000 lb-ft for this system. The attitude-control system has been sized for 60,000 lb-sec impulse to provide a 20 percent reserve.

The re-entry attitude-control requirement is primarily one of roll control. Since a single flap is used for navigation during this phase of the mission, the vehicle must be oriented properly to correctly position the flap. The torque necessary to rotate the vehicle in the time allowed has been computed to be 120 lb-ft. Again employing couples, the maximum separation possible on the re-entry vehicle is 6 ft, thus requiring a thrust of 20 lb per engine (two engines per couple). The impulse necessary to accomplish the total orientation maneuver is 690 lb-sec. The roll control system must also provide dead band operation and counteract any roll resulting from misalignment of the flap. Total impulse based on a nominal misalignment torque has been computed to be 5000 lb-sec. Therefore, the total impulse the re-entry roll control must supply is approximately 7000 lb-sec.

Information for this report has been derived from technical studies supplied by the following companies:

Thiokol Chemical Corporation, Reaction Motors Division

Aerojet General Corporation

Bell Aerosystems Company

The Marquardt Corporation

6.2 ATTITUDE CONTROL SYSTEMS DESCRIPTION

6.2.1 Attitude Control System Description

The attitude control system for APOLLO is a storable bi-propellant system employing 12 attitude control engines arranged to form six couples; four of 110 lb-ft and two of 140 lb-ft. The propellants chosen are nitrogen tetroxide and 65 percent hydrazine/ 35 percent monomethylhydrazine mix which will be maintained at 150 psia for the duration of the mission. The pressurizing agent for the propellant tank is nitrogen stored initially at 3250 psia in a high-pressure tank. The tank will be charged prior to launch and will pressurize the system at that time. Pressurization should be done on the pad, so that if some serious malfunction occurs at or during launch, the vehicle and personnel can still be returned safely. The gas tank will be spherical in shape and constructed of titanium for high strength and light weight. The initial stress conditions are the most severe, since as the propellant is consumed, the gas expansion in the sphere will reduce the nitrogen from 3250 psia to 300 psia. The total weight of pressurizing gas is too small to make gas heating practical, so that expansion in the cold state will take place. For this "cold expansion" mode of operation, nitrogen with its low ratio of specific heats gives the best performance. Helium or other lightweight gases do not offer a significant weight advantage and suffer from greater leak rates.

To reduce the high storage pressure to the 150 psia propellant tank pressure, a pressure regulator is necessary. For safety and reliability, redundant pressure regulators are recommended. The complexity associated with redundant regulators can be reduced by having one regulator serve as a non-operating standby, isolated by a normally closed squib valve. Should the propellant tank pressure rise above or fall below the desired limits, a simple failure-sensing device can isolate the operating regulator by firing the normally opened squib valve, and activate the non-operating regulator by firing the normally closed squib valve. To guard against the open failure of both regulators, a pressure-relief valve will be incorporated on or near the propellant tanks.

In each pressurizing line, a check valve will be installed to prevent propellant vapors from coming in contact. Single valves are adequate since the probability of vapors being present and capable of flowing to the other tank is slight. Failure of the check valve would be in the open position, so no serious pressurization consequences would result.

The two propellant tanks will be made of lightweight aluminum or titanium alloys which are compatible with the propellants. The propellants will be sealed within them either by a squib valve or a diaphragm and will not be released until gas pressurization of the tanks occurs. This storage method improves handling and is a personnel safety feature. For the quantity of propellants necessary, individual tanks will be employed. Multiple tanks add to the weight and reduce reliability (increased leakage, etc), because of the necessity of cross-manifolding. If cross manifolding is not employed, either two completely separate attitude control systems are necessary, or each tank must feed half the total number of attitude control engines. Both arragements are undesirable from the weight standpoint; the former requiring complete redundancy, the latter requiring redundant propellant. Redundant propellant would be necessary because failure of one engine means the total impulse for that axis must be supplied from one tank rather than split equally between both tanks. Thus, each tank must carry sufficient propellant for the entire mission instead of merely half. The two propellant tanks will be placed on opposite sides of the longitudinal axis, and at a distance commensurate with their relative weights to preserve the symmetry of the vehicle.

The propellants are separated from the pressurizing gas by a flexible bladder. Current bladders in use are Teflon, but recent Advent tests indicate Teslar to be a superior material in a radiation environment. Lightweight Teslar bladders can be made 0.004 in. thick and still have sufficient Nitrogen permeability resistance to maintain the propellant and pressurizing gas isolation for the length of the APOLLO mission. These flexible bladders can be tested before service as opposed to metal bladders thus allowing detection of imperfections and faults which could cause failure. Expulsion efficiencies for the tanks using hemispherical bladders would be in excess of 99 percent. The hemispherical bladders would be blown inside out during expulsion, thereby allowing full utilization of the tank for propellant storage.

Each propellant will be transmitted to the attitude control engines by a single manifold. This manifold extends from the propellant tank to the aft end of the vehicle and then around the vehicle's periphery but always remaining with the fairing. Variations of flow rate and pressure within the manifold will be negligible as most of the pressure drop for each engine occurs across the calibrating orifice and injector. The manifold will be of moderate diameter (approximately 1/2 in.) serving as a plenum and surge chamber (to keep the hydraulic ram effect from becoming serious), and acting as a heat sink to maintain fluid temperature. Additional thermal stability will be provided by insulation for the entire manifold length, and by heating units installed at critical locations. The manifold, as well as the small lines to each engine will be of an aluminum alloy which is compatible with both oxidizer and fuel. The propellant tanks and manifold will be filled prior to launch after a Nitrogen purge and evacuation of the entire system. To prevent accidental clogging of the engine injector and valves, 5-micron filters will be included in the manifold.

The engines will be radiation cooled engines of 6 or 8 lb thrust each, located at the periphery of the aft end of the vehicle. They will be grouped in two sets of four and two sets of two arranged in opposition (a separation of 18 ft per couple) to preserve the vehicle symmetry. Each engine will consist of a chamber and nozzle, two quick-response propellant valves, and calibrating orifices to maintain the thrust properly. The expansion ratio was selected at 40/1. The chambers and nozzles will be radiation cooled and constructed of a high-temperature molybdenum of tungsten alloy. An internal coating to reduce heat transfer and a high emissivity coating to keep the wall temperature from rising too high may also be employed. The expected equilibrium temperature should be 2000 to 3000F. If conditions at the throat are severe, some high-melting refractory material such as pyrolitic graphite may be necessary. Experience in the Advent program has shown that thermal feedback from the engine to the hydrazine after shutdown is negligible, and a temperature rise of no more than 35 degrees is anticipated. Ablation cooling has not been recommended because of the weight penalty. The ablation material when designed for the worst conditions (which for APOLLO are a maximum single impulse bit of 200 lb-sec and 3000 starts for smaller impulses) results in a heavier engine.

The propellant valves will be high-response solenoid valves which are normally closed. These valves require an electrical signal to open and remain open. Therefore, failure of this type of valve is primarily in the closed position. Failure in the open position would be the result of corrosion in the valve or a small particle preventing the poppet from seating. Care will be taken to insure compatibility of the valve material with the propellants, and filters should remove stray particles. As a safeguard, normally open latch-type solenoid valves would be placed at each of the four engine locations to shut down the group if one of the engines remains in the open position. When propulsion is needed, the valve would open allowing propellant to flow. These valves would be normally open only during the 14-day mission time. At launch, the valves would be closed, necessitating two separate events before firing as prescribed by the pad safety sequence requirements. While the operation of these valves during flight can be automatic, the pilot should be notified of a failure of any of the engines. This could be accomplished by having thrust transducers on each engine connected to a visual display in the cockpit. The engines themselves have an on-off mode of operation with a short response time. The response time expected for engines of this size would be 15 to 20 milliseconds, the exact value depending upon the characteristic length of the engine and the electrical input to the valves. As an example, the current response time for the 25-lb thrust engine used on Advent is 15 milliseconds with a characteristic length of 10 in., and a surge current of one amp. Proportional engines offer no advantage with this amount of thrust, and tend to increase the control system complexity. Since the limit cycle operation for APOLLO is determined by the angular rate sensitivity of the gyros, rather than the response of the engines, there is no propellant savings associated with a proportional control.

A schematic of the attitude control system is shown in Figure I-6-1 and a weight estimate may be found in Table I-6-1.

6.2.2 Re-entry Roll Control System Description

Roll control on the re-entry vehicle is to be supplied by four 20-lb thrust engines arranged in two opposing couples of 110 lb-ft torque each. The requirements for the re-entry roll control are similar to those of the main attitude control. Monopropellants are comparable to bi-propellants, but do not lend themselves to long-term storage or quick response. Consideration of these results indicated the best system to be the basic arrangement used on the midcourse attitude control system.

Nitrogen gas at 3250 psia will be reduced to the operating pressure by means of redundant pressure regulators. The gas will be contained at 3250 psia until re-entry occurs, at which time a squib valve will fire to pressurize the propellant tanks. The propellants will be the same ones employed in the main attitude control system. They will, however, be sealed within their tanks by a metal diaphragm. A diaphragm is less susceptible to leaks than valves, a feature which is essential because of the proximity of the tanks to the crew. The propellant lines are quite short (approximately 5 ft) and are sufficiently close to the cabin environment to eliminate the need for large amounts of insulation. Filters will be employed in these lines to prevent clogging.

The engines themselves will be located external to the vehicle in two groups. The expansion ratio of the nozzle will depend upon the altitude at which roll control is necessary. An expansion ratio of 10/1 appears probable. The high temperatures encountered in re-entry may prevent radiation cooling from being employed. In this case, ablation cooling may be necessary.

Roll control alone has been stipulated, however, pitch and yaw may be required depending upon the re-entry method. Pitch and yaw control would be required primarily at the start of re-entry, and the anticipated total impulse should be approximately 10 percent of roll. The pitch and yaw control engines would operate in the same manner as the roll control engines. A schematic of the re-entry system may be found in Figure I-6-2 of the accompanying diagrams, and a weight estimate is given in Table I-6-1.

TABLE I-6-I. ATTITUDE CONTROL SYSTEM WEIGHT COMPARISON

ITEM				
Propellants	N_2H_4 MIX - N_2O_4		LH_2 - LO_2	
Specific Impulse, sec	300		425	
Operating Pressure, psi	150		150	
O/F	1.48		4.30	
Total Impulse, lb-sec	7000 (Re-Entry)	62,000 (Main)	7000 (Re-Entry)	62,000 (Main)
Propellant Weight, lb	23.40	208.10	16.6	146.7
Oxidizer Weight, lb	14.0	124.1	13.46	119.0
Fuel Weight, lb	9.4	84.0	3.14	27.7
N_2 Weight, lb	.25	2.03		
H_e Weight, lb			1.49	13.10
Pressure Gas Tank Assembly, lb	.41	3.61	1.96	17.23
Pressure Regulator, lb	.75	.75	.75	.75
Pressure Relief Valves, lb	4.00	3.00	4.00	3.00
Squib Valves, lb	.76	.76	.76	.76
Filters, lb	.24	.24	.24	.24
Lines & Fittings, lb	1.40	1.40	2.50	2.50
Propellant to Fill Lines, lb	.30	.30	.20	.20
Propellant Tank Weight, lb	1.81	16.65	3.91	34.50
Total Weight, lb	33.82	237.34	32.91	219.48
Oxidizer Tank Volume, cu. in. (With Ullage)	300	2630	365	3,238
Fuel Tank Volume, cu. in.	300	2630	1,285	11,322

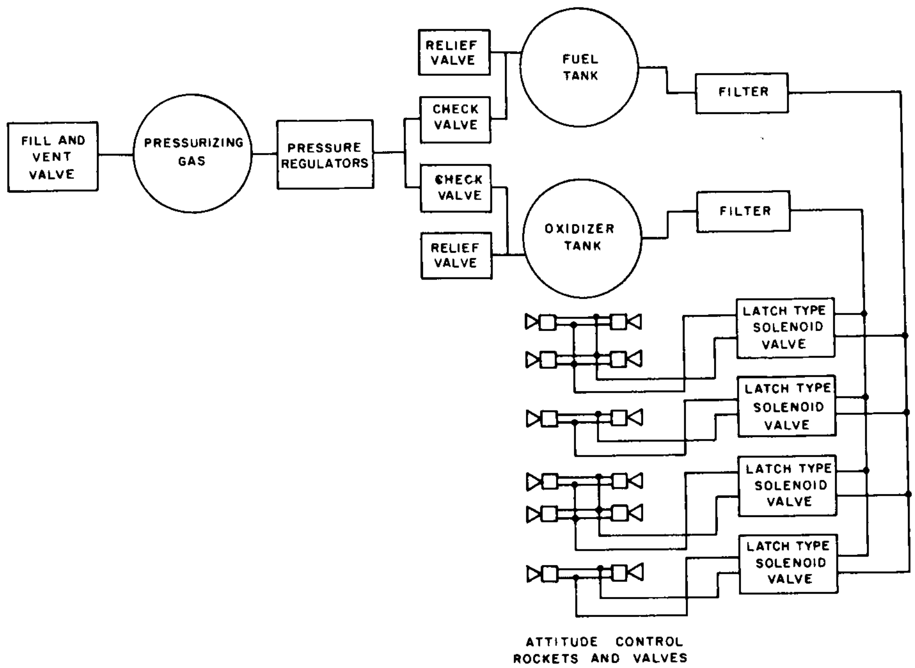


Figure I-6-1. Main vehicle attitude control system schematic

6.3 ATTITUDE CONTROL SYSTEM SELECTION

6.3.1 Operational Mode Selection

To change orientation in space, a torque must be applied to initiate vehicle movement, followed by a counter-torque to stop the vehicle at the proper position. This torque may be applied by a couple (two forces acting in opposite directions and separated by a fixed distance), a single force (acting at a distance from the center of mass and not passing through it), or by changing the momentum of a flywheel. Momentum flywheels offer great advantage when the orientation of the vehicle must be held within very narrow limits and the disturbing torques are almost negligible. With the present APOLLO navigation system, narrow orientation limits are unnecessary, and the disturbing torques (due to nonuniform gravitational field, solar pressure, internal movement with the vehicle, etc.) would be too large for a flywheel alone to control. The additional requirement that the vehicle be rotated rapidly further reduces the attractiveness of

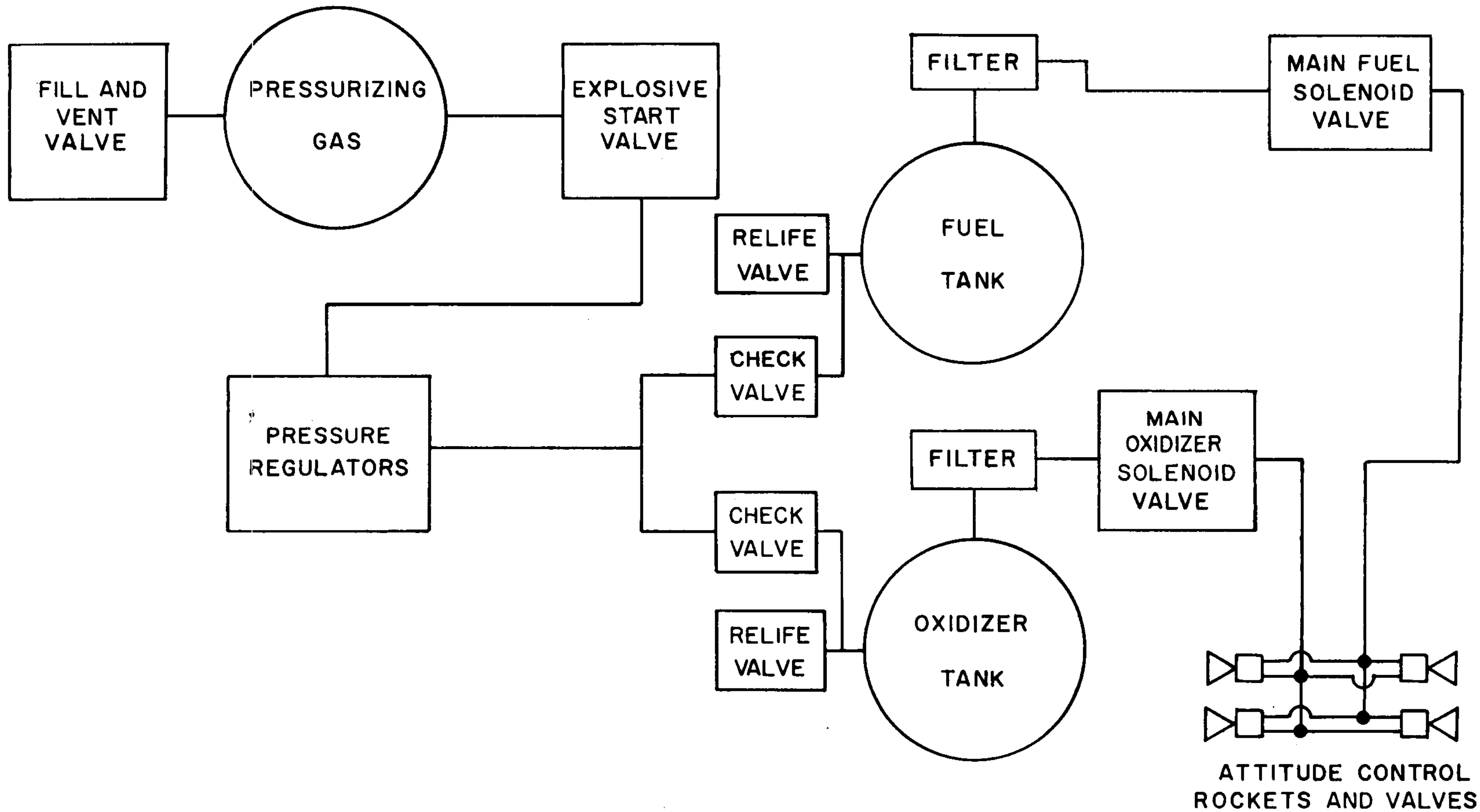


Figure I-6-2. Re-entry vehicle attitude control system schematic mementum flywheels. A combination of momentum flywheels and a reaction system would satisfy all conditions, but offers no advantage for the APOLLO vehicle based on current environmental definitions. Should these definitions change and flywheels be desirable, a reaction system could be employed to perform the major navigational manuvers and de-satureat the flywheels when necessary. The response and impulse capability of the present system is more than adequate to accurately perform these tasks.

Consideration of the effects of the two reaction methods of attitude control and the vehicle characteristics themselves indicate that couples give the best performance. This is attributable to several factors the most dominant one being non-perturbation of the trajectory (or orbit) by couples. A couple has no translative force, and produces a torque only, whereas a single force would produce not only a torque, but would translate the vehicle along the direction the force was acting. In limit-cycle operation, torque must be applied frequently. Hence, to maintain the desired trajectory, additional midcourse propulsion would have to be expended to compensate for the errors introduced by the attitude-control system. An additional factor which has been given consideration is the effect of the location of the center of mass on the applied torque. A single force produces torque as a result of acting at a distance from the center of mass. Since the center of mass varies because of propellant consumption, the moment arm changes and, hence, the torque changes. A couple, however, is insensitive to the location of center of mass and will produce the same torque throughout the mission. This constancy of torque is advantageous from the control standpoint as variation of the moment of inertia and variation of the torque makes exact orientation of the vehicle difficult.

From the reliability standpoint, couples represent the best combination of redundancy and weight optimization. The failure of one engine in the closed position (the open position is unlikely and has been considered previously) eliminates the couple, but torque is still available from the remaining engine. With the present vehicle configuration, this torque would be approximately one-half that of the couple. From the attitude control standpoint, the consequences of reduced torque are not serious as the remaining engine will operate for twice the normal firing time to give the vehicle the desired angular impulse. Since the attitude control system is used primarily for orientation and not to counteract any thrust misalignment torques, the problem of maintaining attitude during thrusting does not exist. The stray torques (solar pressure, etc.) are expected to fall below the 55 lb-ft of torque available, so the vehicle will still maintain attitude. With the couple effect eliminated, trajectory perturbations will result. The perturbations should not be as severe as would be encountered in a system utilizing single forces, since the thrust is halved.

6.3.2 Attitude Control Propellant Selection

In choosing the proper propellant for attitude control, careful consideration must be given to the mode of operation of the attitude-control system itself. Multiple restarts, quick response and high performance are the primary requirements. In addition, the capability of delivering short accurate impulses is essential. With these criteria considered, weight calculations based on various system have been made. For a cold gas system, the total weight of the gas and its tankage, based on the 60,000 lb-sec total impulse, will exceed 2000 lb. This represents an insurmountable weight handicap, so a gas system has been discarded for use with the APOLLO vehicle. Existing monopropellants show considerable improvement in performance and weight over cold gas, but do not have the capabilities of present bi-propellant systems. Although the simplicity and reliability of a monopropellant system are superior to bi-propellant systems, the improvement does not offset the weight handicap (approximately 100 lb) associated with their use. Advanced monopropellants have a performance which is almost comparable to storable bi-propellants. Difficulties with many of these propellants (notably Cavea B) indicate that considerable effort must be expended on them before they can be considered for manned applications.

Applying the restrictions to the choice of bi-propellants, a combination that is hypergolic is most desirable. Since a hypergolic combination ignites on contact, quick response is obtained and multiple restarts can be achieved without an igniter. Igniters are undesirable as they increase complexity by requiring a control system programmed to heat each igniter prior to propellant injection. A faulty or delayed ignition (a phenomenon which does not occur with hypergolic propellants) would allow propellant accumulation in the chamber, with resulting catastrophic failure when ignition does occur.

The propellant combination recommended for the main APOLLO propulsion (LH_2/LO_2) does not satisfy this hypergolic restriction and, in addition, both propellants are cryogenic. Cryogenic propellants are difficult to transmit in feed lines because of evaporation. Propellant loss is less of a problem in this respect than is the creation of a two-phase mixture. Firing the engines with this mixture present would cause inefficient and unpredictable combustion, resulting in an erratic and uncertain torque output. Insulation of the lines would reduce this evaporation but cannot completely eliminate it. An active temperature control to maintain the low temperatures would require a refrigeration cycle of a high level and would consequently have high power requirements and weight penalties. Storable propellants would not suffer from evaporation but may require a small heating device at critical locations to keep the propellants from freezing. Since heating is simpler than refrigeration, the temperature control for storables is less complex than for cryogenics.

Acquisition is another difficulty associated with cryogenics. Acquistion for the main APOLLO propulsion is accomplished by firing small rockets (ullage rockets) prior to each thrusting maneuver to move the propellants to the outlets of their respective tanks. Attempting to use this sequence for attitude-control acquisition would require a large amount of propellant, as the ullage rockets would have to fire prior to each attitude control thrust. Consequently, some alternate method of acquiring the propellants is necessary. For storable propellants this method consists of storing the propellant in a flexible bladder and expelling it by gas pressure.

Current storable propellants offer the best avenue of approach to the attitude control propellant selection. A study of the propellants shows the best over-all characteristics to be possessed by the combination of nitrogen tetroxide and a 65 percent hydrazine-35 percent monomethylhydrazine mi. The particular hydrazine mix chosen has almost the same freezing point as nitrogen tetroxide, thereby increasing the operation temperature range. An additional factor is that with an oxidizer-fuel ratio of 1.48, the tanks are of equal size, thereby reducing development and manufacturing costs. The card gap sensitivity of hydrazine is known to be approximately four and will be reduced by the addition of the more stable monomethylhydrazine to acceptable limits. This propellant combination is also one on the higher performing storable combinations available and General Electric MSVD, as well as other companies, has accumulated vast experience and knowledge with comparable hydrazine combinations. Consequently, the high degree of reliability and safety that can be obtained through long experience and use, may be achieved. The propellants ability to be completely sealed and isolated is valuable for the main attitude control, and essential for the re-entry attitude control where the tanks are in close proximity to the crew.

A comparison of propellant characteristics (response, development costs, etc.) can be found in Table I-6-II.

6.4 MANNED SAFETY

For a manned vehicle, the concept of safety includes not only a fail-safe mode of operation but some capability utilization after failure. For components systems which are critical, the concept places a requirement for redundancy. Complete redundancy of every component and system is obviously not practical, hence, reliability becomes the dominant factor. Safety in the APOLLO attitude-control system is accomplished by employing proven hardware, and by redundancy where this high reliability cannot be achieved. Additional safety is provided by redundancy of extremely critical components.

The system itself is designed to provide the maximum assurance of success and minimize the possibility of catastrophic or total failure. Well-known and tested propellants have been chosen which, by their nature (hypergolicity), eliminate several modes of failure. Redundancy has been employed where operation is critical (i.e., pressure regulators and engines) especially for the components of inherently lower reliability. Additional safeguards against failure such as pressure reliefs and latch-type solenoid valves have been incorporated.

As a ground safety feature, the pressurizing gas sphere is left totally unpressurized until immediately prior to launch. To guard against accidental firing, two distinct events must occur before attitude-control thrust is possible. As a further guard against injury to the ground crew and occupants, the propellants for the main attitude

TABLE I-6-II. PROPELLANT CHARACTERISTICS COMPARISON

Item	Reliability	Total System Weight, lb	Development Cost (Including Prime & Qual.)	Response	Impulse Accuracy	Development Time
Stored Gas	Excellent	3340	$ $.1 \times 10^6$	5-10 msec	Excellent	1 - 3 months
Monopropellants	Good	330	$ $.5 \times 10^6$	10-50 msec	Fair	6 months
Storable Bi-Propellants	Good	237	$ 3.7×10^6	10-20 msec	Good	12 months
Cryogenic Bi-Propellants	*Poor	219	$ 5.5×10^6	*Poor	Good	18-24 months
Hybrids	*Poor	250	$ 5.5×10^6	*Poor	*Poor	18-24 months

* No Data Available

control and re-entry attitude control are completely sealed within their respective tanks, thereby reducing the possibility of leakage. Each propellant is maintained in a separate tank to prevent explosions or fire if some leakage should occur.

7.0 Lunar Landing

A preliminary analysis of the propulsion requirements for a vehicle capable of a soft lunar landing and a return to earth has been made. The assumed requirements are given below.

Payload	=	10,780 lb
Δ V, Soft Landing	=	9500 ft/sec
Δ V, Return	=	9500 ft/sec
Δ V, Midcourse	=	250 ft/sec (each way)
	=	500 ft/sec (total)
Total Mission Δ V	=	19,500 ft/sec
Fuel Reserves	=	5 percent

Calculations were made for a liquid hydrogen/liquid oxygen propulsion system and for a nuclear system. For the chemical system, it was assumed that the powerplant and empty tankage used for soft landing were left at the lunar surface. A second powerplant was utilized for the return flight.

Data for the nuclear system were provided by the Aircraft Nuclear Products Department of the General Electric Company. The assumption was made that the empty tankage required for the landing portion of the flight was discarded prior to return. However, the same engine was used for both segments of the flight. A specific impulse of 800 seconds was assumed. The weight of the engine and shielding was estimated to be 5000 lb.

The results of the analyses are given in Table I-7-I. For the chemical system, the total weight is approximately 90,000 lb, while the comparable value for the nuclear system is 51,000 lb. It may be noted that the lunar launch weights for the return portion of the flight do not show as large a difference as the total vehicle weight. This may be attributed to the high weight required for the nuclear engine and associated shielding.

TABLE I-7-I

LUNAR LANDING AND TAKE OFF, Δ V = 19,500 FPS
ASSUMES OUTBOUND TANKAGE & ENGINES LEFT ON MOON

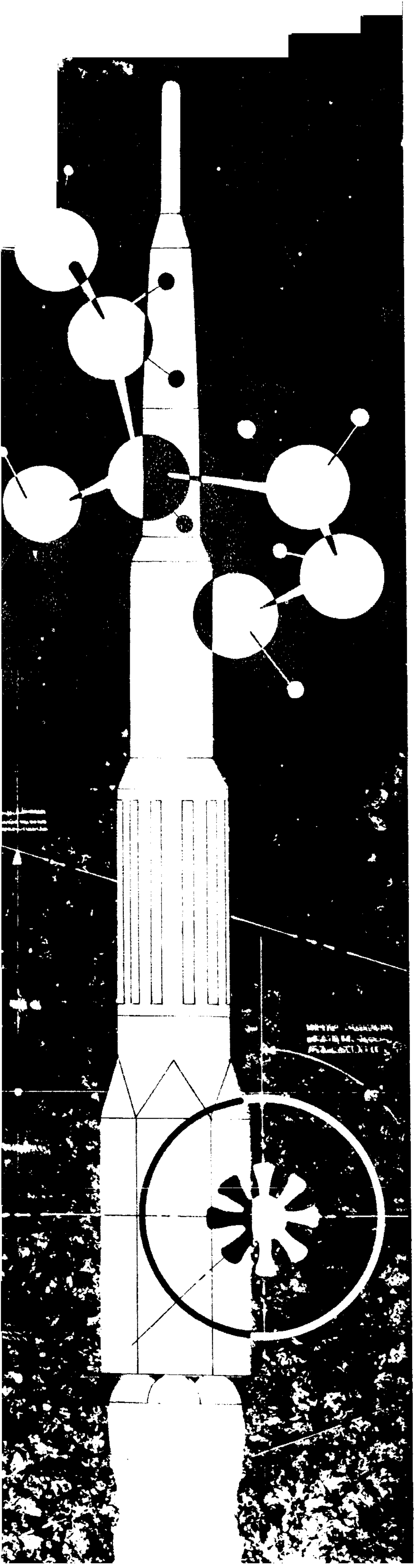
5% Fuel Reserve		
	H_2/O_2	Nuclear*
Return Flight		
Payload	10,780	10,780
Attitude Control	300	300
Propulsion Weight	19,920	14,520
Lunar Launch Weight	31,000	25,600
Outbound Flight		
Payload, lb	31,000	25,600
Attitude Control	800	600
Propulsion Weight, lb	55,200	21,300
Weight at Escape Velocity, lb	87,000	47,500
Abort Rocket Weight, lb	2,200	2,200
Separation Rockets Weight, lb	---	---
Adapter Weight (est.), lb	1,200	1,200
Total Weight on Pad, lb	90,400	50,900

* I_{sp} = 800 seconds (Data estimated by Aircraft Nuclear Products Department of General Electric Company)

The LH_2/LO_2 system for the return flight is in the same size category as the D-2X vehicle outlined previously in Table I-4-VI. Thus, the return vehicle propulsion represents a growth (mainly in tankage) from that recommended for the D-2 vehicle. For the outbound flight, it is possible that a de-rated J-2 engine might find application, although further study is needed.

It is presumed here that the nuclear powerplant is capable of shutdown and restart for return to earth. This presumes that currently proposed methods of shutting off nuclear powerplants without detrimental residual heating can be achieved – an assumption yet to be proven. However, if this nuclear restart proves unfeasible, it would be possible to return with conventional combustion of H_2 and O_2 without serious loss of payload.

From the total weights given in Table I-7-I, it is quite apparent that a booster beyond the Saturn C-2 will be necessary to accomplish a lunar landing with a single earth landed system. By using several C-2 boosters, however, it may be possible to assemble a lunar landing vehicle in orbit. Again, this is an area that will require additional study.



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15 March 1961

REPORT NO. SR-60514-5

Project Apollo

ONE DESIGN APPROACH TO A COMBINED SOLID AND LIQUID PROPELLANT PROPULSION SYSTEM

Aerojet-General CORPORATION

SOLID ROCKET PLANT • SACRAMENTO, CALIFORNIA

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I. INTRODUCTION

For the maneuver requirements of the Apollo spacecraft on a lunar orbit mission, several integrated solid and liquid propellant systems are being studied by the Aerojet-General Corporation. One approach to such integrated propulsion systems is described in this report as an example. The system described uses solid-propellant rockets for launch-abort escape and lunar-orbit injection and exit; storable liquid propellants are used for midcourse guidance and attitude control. Further data concerning storable and cryogenic liquid-propellant systems for the lunar orbit maneuvers are being prepared. Although solid propellant motors do not have as high a specific impulse as the high-energy cryogenic liquid propellant systems, they do offer some interesting features with respect to reliability (crew safety and mission completion), utmost compactness, and simplicity.

II. PROPULSION SUBSYSTEMS

The on-board propulsion system consists of the following subsystems:

A. LAUNCH-ABORT ESCAPE

A means of fast separation of the crew module from the booster rocket is provided in case of a malfunction or irregularity during the first-stage and early second-stage operation. After passing successfully through these critical phases, the escape motor is jettisoned. Since the escape motor is not carried up to escape velocity, it only partially counts as payload for the booster rocket.

B. ATTITUDE CONTROL

The vehicle is oriented with respect to space-fixed coordinates.

II. Propulsion Subsystem (cont.)

C. MIDCOURSE CONTROL

Impulses are produced as required to correct the trajectory during transit from earth to the vicinity of the moon and on the way back to earth.

D. ORBIT INJECTION

The injection propulsion produces the required velocity decrement to achieve a lunar orbit.

E. ORBIT EXIT

The orbiting spacecraft is provided with the velocity increment necessary to leave the lunar orbit and return to earth.

The most critical phase of the lunar orbit mission is the exit phase (subsystem E). A failure to achieve the required impulse is equivalent to a loss of the crew and the vehicle.

Also mandatory is the reliable performance of the midcourse control system for a safe mission completion or emergency return.

The injection phase (subsystem D) is considered to be less critical. A failure to produce the desired velocity decrement results in over-shooting and means that the lunar orbit mission has to be abandoned. Using the orbit-exit propulsion, completely or partially, allows transforming the planned trajectory into a circumlunar one with a delayed, but probably safe, return to earth.

III. SELECTION OF SUBSYSTEMS

The emphasis of highest possible reliability in subsystems C and E is reflected in the selection of redundant systems.

The requirements for the escape system, high thrust level and short duration, are best met with solid propellant motors.

Attitude control requires many small and unpredictable pulses and is best accomplished by a hypergolic liquid-propellant system with start-stop capability.

The midcourse control system also may require many pulses at different thrust levels. This requirement of flexibility favors the use of a hypergolic liquid propellant system. Reliability may be achieved by the use of dual thrust chambers and dual sets of fluid controls to provide a redundant subsystem.

The orbit injection system has to deliver a high impulse to a predetermined maximum value in a relatively short time. A solid propellant motor with highly accurate total impulse control by thrust reversal achieves this. The solid propellant rocket has the further advantage of simplicity and, therefore, inherent reliability. Storability in the space environment without special provisions, good performance, and compactness are also inherent qualities of the solid-propellant motors.

A highly reliable orbit exit propulsion can be attained by splitting the total amount of propellant into several highly reliable solid propellant motors. One additional redundant motor can replace any one of the other motors, thus keeping the potential total impulse at the necessary level. Al o, the weight penalty for redundancy is thus kept at a fraction of the total system weight. The redundant motor can be a solid propellant motor or the midcourse-propulsion liquid-propellant system. The integrated solid-liquid configuration is shown in Figure 1

IV. SYSTEM DESCRIPTION

The spacecraft and the selected subsystems are shown in a schematic drawing in Figure 1. To show the application of an integrated propulsion system, a vehicle configuration was assumed which only reflects the basic Apollo idea: command module, mission module, and propulsion module.

The attitude control system consists of eight thrust chambers with possibly eight additional ones as redundancy. Each chamber develops 10 lb of thrust. The attitude control system is not shown in Figure 1; however, the pressurization and tank system are integrated in the midcourse subsystem. The total impulse is 60,000 pound-seconds.

The escape motor is connected to the command module by means of a Mercury-type tower structure.

In form of two half-rings, the mission module is located around the command module.

In the case of a launch abort, the mission module and the fairing are separated and jettisoned sideways, and the propulsion module remains on top of the launch vehicle. The command module is now free and can be rapidly lifted by the escape motor to a safe altitude to deploy the descent mechanism. With a command module weight of approximately 7000 lb, the 1KS-130,000 motor shown can achieve an altitude of 300 ft in the first second, and the module will coast to an apogee altitude of 4000 ft within 15 sec. The maximum acceleration is 18 g. These values are only approximate, since the performance depends very much on the drag of the capsule.

IV, System Description (cont.)

In the case of a launch abort at maximum dynamic pressure conditions (approximately 35,000 ft), the escape motor develops a thrust of 140,000 lb and can achieve a separation of the capsule from the booster of approximately 100 feet in 1 second.

An alternative motor design with a forward nozzle arrangement may shorten the tower structure. A detailed description of the escape motor is presented in Aerojet-General Report No. SR-60514-2A.

The single motor for orbit injection propulsion is located at the centerline of the vehicle. A fiber-glass chamber and a fixed light-weight nozzle are used to achieve a high mass fraction. To compensate for initial thrust misalignment, shifts in location of the vehicle center of gravity, and shifts in thrust vector during burning, provisions are made to vector the motor. The motor is mounted on three ball-nut screw-jack actuators which allow a 1.5-degree inclination of its axis in all directions.

This orbit-injection motor is equipped with a thrust-reversal device to terminate the thrust at the moment the desired velocity increment is achieved or to terminate the thrust immediately after an improper firing The thrust termination is achieved within 1 to 3 milliseconds after command. At the moment of thrust reversal, the motor is disconnected from the spacecraft by explosive bolts and jettisoned.

The exit propulsion system shown in Figure 1 consists of 10 motors placed around the injection motor. The motors have fixed nozzles, are rigidly mounted, and are adjusted in such a way that the thrust vectors intersect in one point on the vehicle axis.

IV. System Description (cont.)

Since the exit propulsion motors are of the type to achieve high thrust and short duration, they are not fired simultaneously. The firing schedule provides for firing the first six motors simultaneously, thrust termination, explosive disconnect, and jettisoning of the six motor casings. Then, in the second pulse, the remaining four motors are fired; thrust is terminated after the desired velocity increment is achieved, and the motors are jettisoned.

If one of the six motors or one of the four motors fails to ignite, one of the redundant liquid-propellant midcourse engines is lighted and aligned so that the resultant thrust vector points through the center of gravity of the spacecraft. Therefore, no turning moment exists. With the attitude control, the vector is brought into flight direction. If one midcourse engine fails to light, the other engine would automatically start and replace it.

The location of the two gimbaled thrust chambers is also shown in Figure 1, along with tanks for fuel and oxidizer and for the pressurization helium of this dual-purpose liquid midcourse propulsion subsystem. A more detailed description, system diagrams, and reliability analysis, are presented in Aerojet-General Report LRP-PDR 61-5, "General Mechanization Scheme and Reliability Analysis for Project Apollo," dated 27 January 1961.

V. WEIGHT AND PERFORMANCE

For this study, a vehicle weight of 7000 lb without propulsion was assumed. A velocity increment of 250 ft/sec for midcourse correction for each way and a velocity increment of 3150 ft/sec for the injection and exit maneuver were also assumed.

The tables presented at the end of this report show preliminary data pertaining to the Apollo vehicle, in general, and propulsion systems. A weight tabulation for significant points on the trajectory is shown in Table 1. Performance and weight data for the launch-abort escape motor are shown in

V, Weight and Performance (cont.)

Table 2. For the midcourse-guidance liquid propellant system, data are shown in Table 3 to supplement references in the text of this report. Data concerning the solid propellant motors used for orbit injection and exit are presented in Tables 4 and 5, respectively.

Table 1

WEIGHT OF APOLLO VEHICLE AT SIGNIFICANT POINTS IN TRAJECTORY

At:	Vehicle wt lb	Component wt lb	Component Burned-Out or Jettisoned
Launch	16,599		
		765	Escape Motor
Escape velocity	15,834		
		344	Midcourse propellant, earth to moon
		120	Attitude control propellant, earth to moon
Lunar orbit injection	15,370		
		4438	Injection motor
Lunar orbit	10,932		
		3297	Exit propulsion system (10 motors)
Lunar orbit exit	7635		
		250	Redundant exit engine propellant
		175	Midcourse propellant, moon to earth
		70	Attitude control propellant, moon to earth
		100	Midcourse propulsion system hardware
		40	Attitude control hardware
Re-entry	7000		

Table 1 (cont.)

WEIGHT OF APOLLO VEHICLE AT SIGNIFICANT POINTS IN TRAJECTORY

Assumptions:

$I_{s\ (solid)}$ = 305 lbf-sec/lbm, vacuum, ϵ= 30:1 (lunar orbit and exit motors)

$I_{s\ (liquid)}$ = 320 lbf-sec/lbm, vacuum, ϵ= 40:1

Mass fraction: Injection motor = 0.95
Exit propulsion system (10 motors) = 0.91

Velocity increment: Midcourse (each way) = 250 fps
Orbit injection = 3150 fps
Orbit exit = 3150 fps

Total impulse of attitude control = 60,000 lb-sec

Table 2

LAUNCH-ABORT PROPULSION SYSTEM

IKS-130,000
SINGLE SOLID ROCKET MOTOR, THREE CANTED NOZZLES

Total weight = 747 lb
Mass fraction = 0.762
Expansion ratio (ϵ) = 18:1
Chamber pressure = 1100 psi
Chamber material - Nickel-steel
Burning time = 1 sec
I_s at altitude (35,000 ft) = 276 lbf-sec/lbm
I_s at sea level = 250 lbf-sec/lbm
Thrust at maximum acceleration (35,000 ft) = 140,000 lb
Thrust at sea level = 127,000 lb

Propellant Properties

ANP-2913 CD: 68% NH_4ClO_4 16% Al
0.3% Ballistic additive
15.7 wt% Polyurethane Binder

I_s at 1000 psi (sea level) = 247 lbf-sec/lbm, measured
Burning rate = 0.7 in./sec at 1000 psi
T_c = 5740°F
Density = 0.064 lb/cu in.

Table 3

MIDCOURSE CONTROL SYSTEM

Liquid: Pressure Feed System with Dual Thrust Chambers and Controls

Propellant: N_2O_4 and 0.5 N_2H_4 + 0.5 UDMH Misture Ratio MR 2.1:1

I_s =	320 lbf-sec/lbm
Expansion ratio =	40:1
Throat diameter =	3.2 in.
Exit cone diameter =	20 in.
Over-all length of thrust chamber =	35 in.
Chamber pressure =	134 psi
Weight of thrust chambers, control module, and tanks ≈	75 lb
Thrust, each engine =	1500 lb

Table 4

ORBIT INJECTION MOTOR

Single Solid Rocket Motor, One Fixed Nozzle

Total weight =	4360 lb
Propellant weight =	4142 lb
Inert weight =	218 lb
Mass fraction =	0.95
Thrust =	31,500 lb
Burning time =	40 sec
$I_{s(vac)}$ (ϵ= 30:1) =	305 lbf-sec/lbm
Chamber pressure =	500 psi
Chamber material =	Glass-fiber-resin composite

Propellant Properties

Propellant with Beryllium Additive: 49.4 wt% NH_4ClO_4

13% Be

37.6% Nitropolyurethane binder

I_s at 1000 psi (sea level) = 264 lbf-sec/lbm, expected measured

T_c = 6000°F

Density = 0.061 lb/cu in.

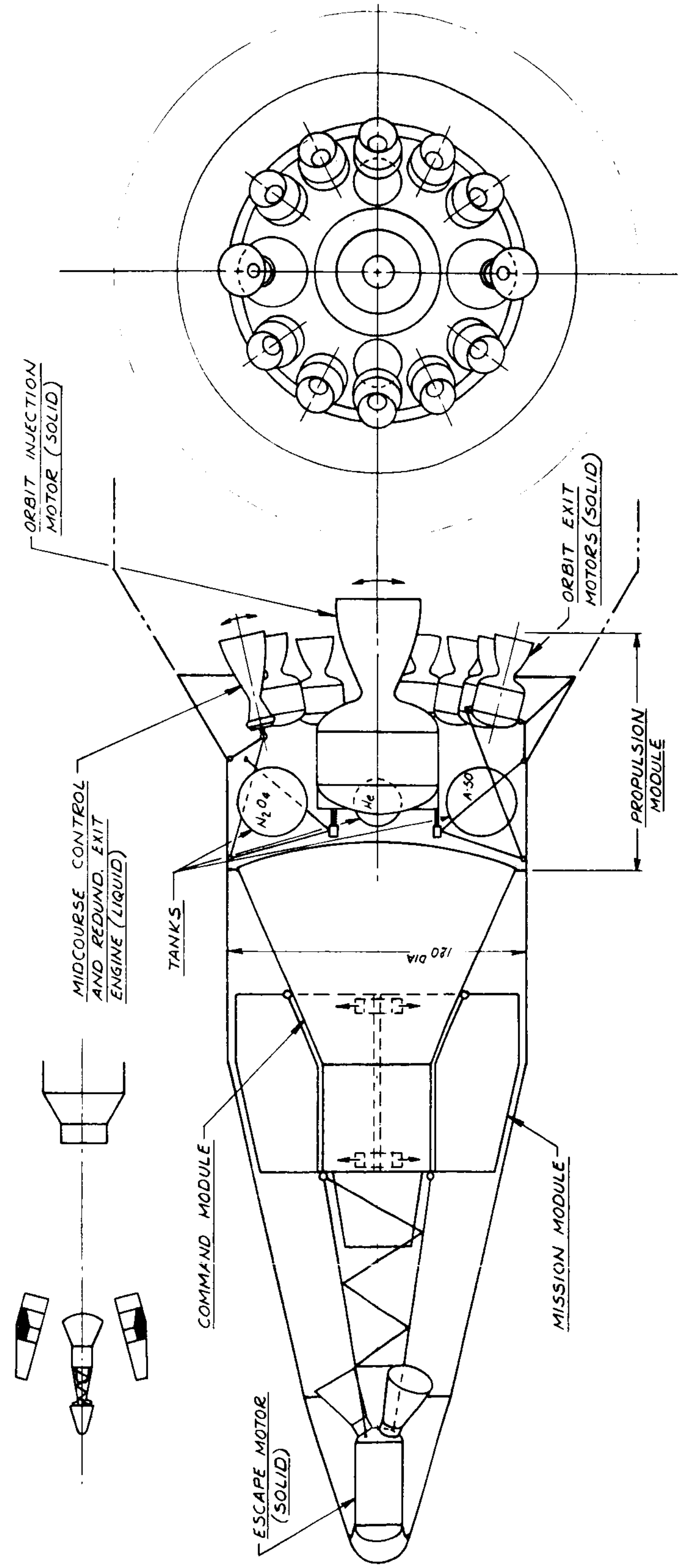
Table 5

ORBIT EXIT PROPULSION SYSTEM

Ten Solid Rocket Motors, Each With a Single Fixed Nozzle

Total weight (each motor) =	324 lb
Propellant weight =	295 lb
Inert weight =	29 lb
Mass fraction =	0.91
Thrust =	3600 lb
Burning time =	24 sec
$I_{s(vac)}$ (ϵ= 30:1) =	305 lbf-sec/lbm
Chamber pressure =	500 psi
Chamber material =	Glass-fiber-resin composite

Propellant: Same as in Table 4



Apollo Solid-Liquid Propulsion System

Figure 1